SCIENCE APPLICATIONS INCORPORATED

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ADVANCED PLANNING ACTIVITY

February 1973 - January 1974

SAI-120-M2

For

Planetary Programs Office
National Aeronautics and Space Administration
Washington, D. C.

By

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CONTRACT NO. NASW 2494 - ADVANCED PLANETARY ANALYSES

25 February 1974

CONTENTS

	Page No.
Jupiter Orbiter Performance Comparison - Earth Storable versus Space Storable Retro Propulsion	5
Jupiter Orbiter Performance Depth with Fixed and Expanded MM '71 Retro Propulsion Subsystems	12
1983 Venus and 1986 Uranus/Neptune SEP Missions	18
1989 Venus and 1981/82 Encke Rendezvous SEP Missions	23
1989 Saturn and 1989 Asteroid (METIS) Rendezvous SEP Missions	29
1987 Mercury SEP Mission	34
Space Shuttle and Planetary Missions	37
Pioneer Saturn and Uranus Entry Probe Mission Dates	42
Comet Kohoutek Fly-By Mission Parameters	48
Recovered Tug Earth Escape Performance	52
Titan Atmosphere Workshop	54
Inputs for Electric Propulsion Conference	57
1985 Saturn Orbiter Performance Curves	62
OOS Tug Evaluation	64
Ballistic Rendezvous with Encke 81/82	76
Comet Encke 80 Fly-By - Asteroid Rendezvous Mission	79
Pioneer Mars 1979 Mission Options	90

Advanced Planning Activity

Science Applications, Inc. (SAI) is engaged in a program of advanced study and analysis for the Planetary Programs Office (Code SL) of NASA. The nature of the work is quite varied ranging from prephase A mission studies to short quick response analysis. The tasks performed between 1 February 1973 and 31 January 1974 are summarized in the end-of-year Summary Report (SAI-120-A1).

One of the contract tasks areas is identified as Advanced Planning Activity and embraces a wide range of analysis that is performed for Code SL on an as needed, and usually quick response basis. The output from these analyses is reported to NASA in the form of memoranda, working papers, letter reports and occasionally as a published report. This document is a collation of the output of all the Advanced Planning Activities for the period 1 February 1973 to 31 January 1974. The papers and memoranda are included in their original form and have been neither edited or up-dated. A total of seventeen analyses are reported as summarized in Table 1.

TABLE 1.

SUMMARY OF ADVANCED PLANNING ACTIVITY

Table 1.

SUMMARY OF ADVANCED PLANNING ACTIVITY

TASK	DATES	TASK TITLE	SUBMITTED TO
1.	FEB. '73	Jupiter Orbiter Performance Comparison - Earth Storable versus Space Storable Retro Propulsion	NASA HQ.
2.	FEB. '73	Jupiter Orbiter Performance Depth with Fixed and Expanded MM '71 Retro Propulsion Subsystems	JPL
3.	FEB. - MAY '73	1983 Venus and 1986 Uranus/Neptune SEP Missions	MSFC/RI
4.	FEB. - MAY '73	1989 Venus and 1981/82 Encke Rendezvous SEP Missions	MSFC/RI
5.	MAR. - MAY '73	1989 Saturn and 1989 Asteroid (METIS) Rendezvous SEP Missions	MSFC/RI
6.	MAR. - MAY '73	1987 Mercury SEP Mission	MSFC/RI
7.	MAR. - MAY '73	Space Shuttle and Planetary Missions	NASA HQ.
8.	APRIL - MAY '73	Pioneer Saturn and Uranus Entry Probe Mission Dates	NASA HQ. /OPSAC
9.	MAY '73	Comet Kohoutek Fly-By Mission Parameters	NASA HQ.
10.	JUNE '73	Recovered Tug Earth Escape Performance	MSFC
11.	JULY - NOV. '73	Titan Atmosphere Workshop	ARC /NASA HQ.
12.	OCT. '73	Inputs for Electric Propulsion Conference	NASA HQ.

SUMMARY OF ADVANCED PLANNING ACTIVITY (Cont'd.)

TASK	DATES	TASK TITLE	SUBMITTED TO
13.	OCT. '73	1985 Saturn Orbiter Performance Curves	NASA HQ.
14.	OCT. - DEC. '73	OOS Tug Evaluation	NASA HQ.
15.	NOV. '73	Ballistic Rendezvous with Encke 81/82	NASA HQ.
16.	NOV. '73	Comet Encke 80 Fly-By - Asteroid Rendezvous Mission	NASA HQ.
17.	NOV. - JAN. '73	Pioneer Mars 1979 Mission Options	NASA HQ. / ARC

JUPITER ORBITER PERFORMANCE COMPARISON

EARTH STORABLE Vs SPACE STORABLE
RETRO PROPULSION

February 15, 1973

TO: Dan Herman, Manager A P & T

FROM: John Niehoff, SAI

SUBJECT: Jupiter Orbiter Performance Comparison - Earth Storable versus Space Storable Retro Propulsion

Summary

Orbited payload capability is examined for three Jupiter opportunities – 1980, 1981/82 and 1983. Payload performance is evaluated as a function of flight time to Jupiter using Titan III E/ Centaur/B II and Shuttle/Centaur/B II launch vehicles. A 30-day orbit with periapse at $3R_J$ is assumed in the analysis. It is concluded that space-storable retro propulsion provides from 75 to 100 kg more orbit payload than earth-storable propulsion when combined with the Titan III E/Centaur/B II during the three opportunities examined. Using the Shuttle/Centaur/B II this advantage with space storable propulsion increases to about 150 kg. It is further concluded that the combination of the Titan launch vehicle with an earth-storable retro propulsion system is marginal for MJO missions. The Shuttle launch vehicle has sufficient additional capability to rate MJO missions for the period 1980 - 83 as acceptable with earth storable retro propulsion.

Discussion

The 1980, 1981/82 and 1983 Jupiter launch opportunities were evaluated for orbiter payload performance. The purpose of this analysis was to determine the relative capabilities of earth-storable and space-storable propellants used in the orbiter retro propulsion system. Resultant orbited payload capability is presented as a function of flight time to Jupiter with two different launch vehicles, 1) the Titan III E/Centaur/B II, and 2) the Shuttle/Centaur/B II.

A number of assumptions were applied in the analysis. First launch periods of 21 days were assumed without any DLA constraints. A fixed orbit period

Cont..

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of 30 days with a periapse radius of $3R_J$ was selected for retro impulse requirements. To these requirements a 250 m/sec reserve was added for navigation and orbit maneuver requirements. The retro system sizes were allowed to vary according to scaling relationships in order to fully utilize the earth escape mass capability. The scaling equations used are as follows:

earth storable: $M_s = 1.15\ M_p + 45$ kg,

space storable: $M_s = 1.16\ M_p + 66$ kg,

where M_p is the propellant loading and M_s is the retro system gross (wet) weight. Specific impulse with earth-storable and space-storable propellants was assumed at 283 sec and 385 sec, respectively. The earth-storable parameters are based on MM '71 technology, whereas the space-storable values relate to proposed FLOX-MMH systems which could be developed within the current state-of-the-art.

Plots of net orbited payload (exclusive of all propulsion systems) versus Jupiter flight time are presented in Figures 1 - 3 for the 1980, 1981/82 and 1983 launch opportunities, respectively. Note that there are two graphs on each Figure, one for Titan III E/Centaur/B II and the second for Shuttle/Centaur/B II. In all cases, peak payload performance occurs at flight times between 750 and 850 days to Jupiter. Comparing the opportunities, one sees that 1981/82 yields the most orbited payload. The 1983 opportunity is almost as good, but the 1980 opportunity exhibits a 15% to 20% decrease in capability.

The payload performance with earth-storable propellants is indicated by the solid curves in the Figures. Dashed curves represent the space-storable propellant payload capability. Almost independent of launch opportunity, it can be observed that the assumed space-storable propulsion system adds between 75 and 100 kg orbited payload in the vicinity of 800 day flight times (i.e. peak performance) using the Titan III E/Centaur/B II. Using the Shuttle/Centaur/B II, this payload advantage is increased to about 150 kg under similar transfer conditions.

Perhaps, more important than these advantages, is the fact that earth-storable retro propulsion combined with the Titan III E/Centaur/B II provides a maximum of only 550 kg orbited payload (1981/82 opportunity). This

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would appear to be a marginal amount for an MJO spacecraft exclusive of propulsion. Hence, either a space-storable retro system or the Shuttle/ Centaur/B II launch vehicle are needed propulsion improvements for effective MJO mission capability and mission planning flexibility.

John

JCN/sn

J. C. Niehoff

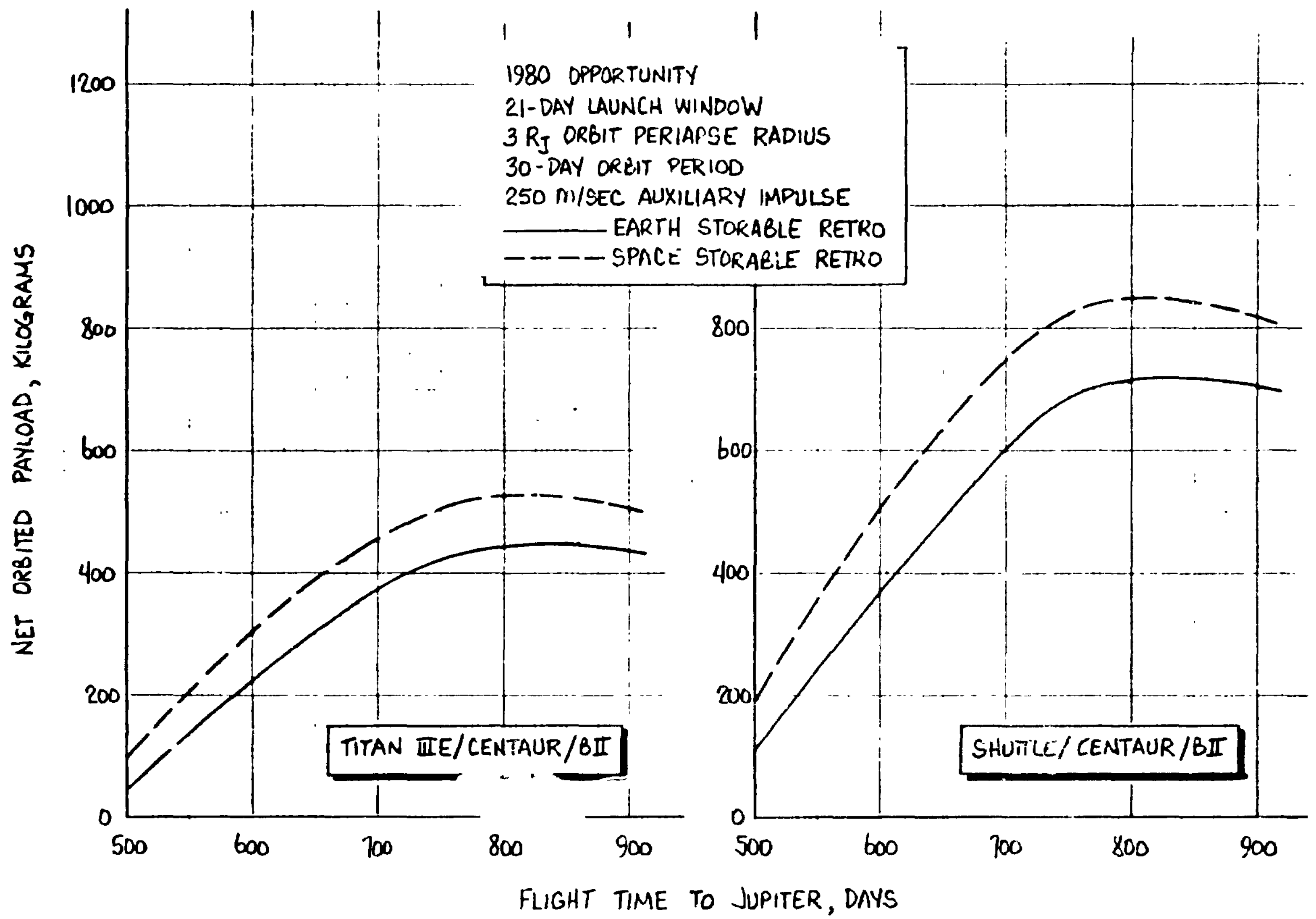

FIGURE 1. 1980 JUPITER ORBITER PAYLOAD PERFORMANCE ($RP = 3R_J$, PERIOD = 30 DAYS)

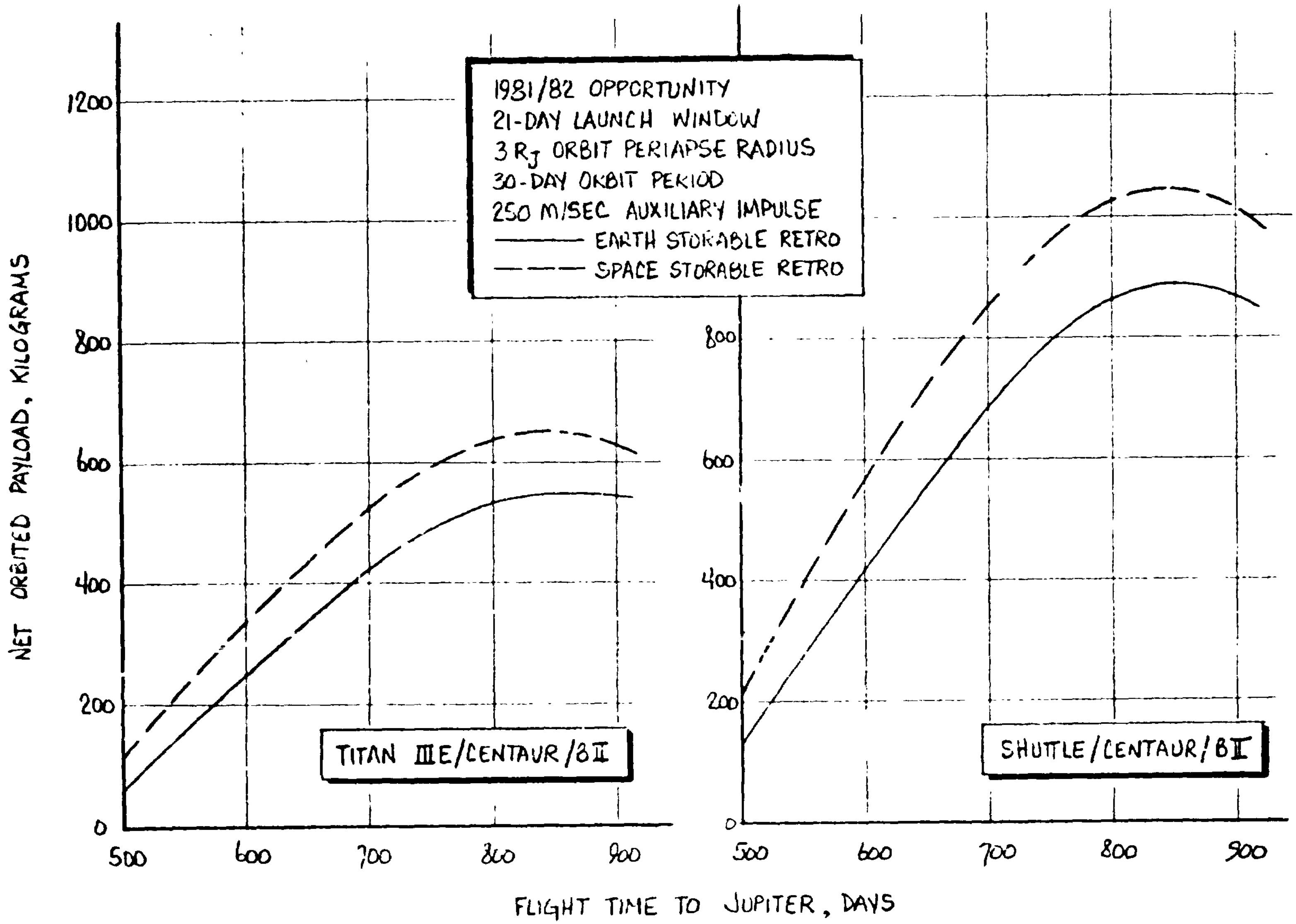

FIGURE 2. 1981/82 JUPITER ORBITER PAYLOAD PERFORMANCE (RP = $3R_J$, PERIOD = 30 DAYS)

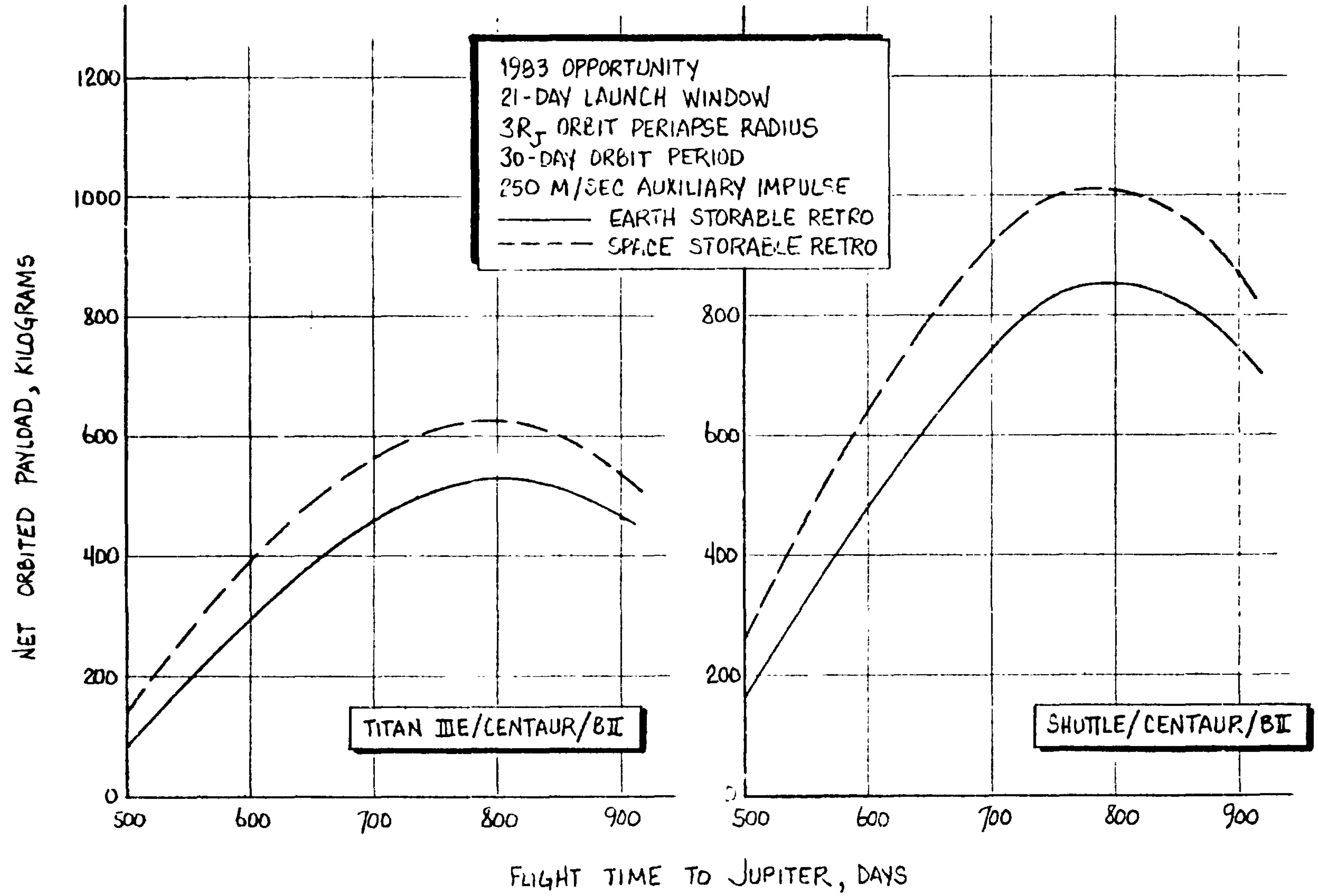

FIGURE 3. 1983 JUPITER ORBITER PAYLOAD PERFORMANCE (RP = $3R_J$, PERIOD = 30 DAYS)

JUPITER ORBITER PERFORMANCE DEPTH

WITH

FIXED AND EXPANDED MM 71 RETRO PROPULSION SUB-SYSTEMS

SCIENCE APPLICATIONS, INC.

February 16, 1973

TO: Jim Long, JPL

FROM: John Niehoff, SAI

SUBJECT: Jupiter Orbiter Performance Depth with Fixed and Expanded MM '71 Retro Propulsion Subsystems

Summary

The total burnout mass capability of a Jupiter orbiter is examined with a MM '71 retro propulsion subsystem. The analysis is restricted to an 800-day mission launched during the 1981/82 Jupiter opportunity. Both the Titan IIIE/Centaur/BII and Shuttle/Centaur/BII launch vehicles are considered. The purpose of this analysis was to investigate the performance depth of the MM '71 retro propulsion subsystem design. Depth of performance is measured by the ability of the retro system to deliver acceptable orbiter burnout mass to a fixed period 30-day orbit with increasing periapse radius. Results are presented which show that less than 600 kg is available for the orbiter (exclusive of the propulsion subsystem) for all orbit periapse radii greater than 2 R_J if the Titan IIIE/Centaur/BII is used for launch. The same conclusion applies to a Shuttle/Centaur/BII launched mission if the propellant capacity is limited by the present MM '71 tank size. However, by increasing the propellant capacity the orbit periapse radius can go as high as 6.75 R_J before the net orbit orbiter mass (excluding the propulsion subsystem) falls below 600 kg. The required propellant capacity at this point would be approximately 2.25 times as large as that of the present design. From this brief analysis it is concluded that acceptable application of the MM '71 retro propulsion system to an MJO mission will almost certainly require expanded propellant capacity. Doubling the tankage, i.e. four tanks instead of two, combined with Shuttle/Centaur/BII launches would provide considerable propulsion flexibility for MJO mission planning.

Discussion

This brief analysis addresses the question of applying the MM '71 earth-storable retro propulsion subsystem to Mariner Jupiter Orbiter (MJO) missions. For this purpose, the minimum energy 1981/82 Jupiter launch opportunity was

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used. Near-maximum payload 800-day trajectories combined with a launch period of 21 days (no DLA constraints) were examined for energy requirements. A maximum C3 of 87 km^2/sec^2 is required for launch and the average Jupiter asymptotic approach velocity is 6.8 km/sec.

Total orbiter burnout masses were computed for a 30-day period orbit with varying periapse radius using the MM '71 retro propulsion subsystem, 1) at its present propellant capacity, and 2) with increased propellant capacity. In its present configuration the total MM '71 propulsion subsystem was assumed to weigh 573 kg of which 449 kg is useful propellant and 124 kg is residual weight. For expanded propellant capacity the total subsystem weight was assumed to vary according to the equation

$$M_S = 1.15\ M_p + 57 \qquad \text{(kg)}$$

where M_S is the subsystem weight, and M_p is the propellant capacity of the tanks. In either of these cases the propulsion specific impulse was assumed to be 283 sec.

MJO payload performance was evaluated with the Titan IIIE/Centaur/BII and the Shuttle/Centaur/BII launch vehicles; the results are presented in Figures 1 and 2, respectively. Considering Figure 1 first (Titan launches), total orbiter burnout mass is plotted as a function of orbit periapse radius. Note that the injected payload capability of the Titan IIIE/Centaur/BII is 1165 kg. Also, a reserve of 250 m/sec impulse for navigation and orbit maneuvers has been factored into the payload calculations. Two total burnout mass curves are presented. The solid curve represents a variable propellant load matched to the total launch capability and orbit ΔV requirement. The dashed curve represents a fixed propellant loading equal to the tank capacity of the present MM '71 retro subsystem. This curve implies that the launch vehicle is off-loaded from its maximum payload capability. At orbit radii less than 2 R_J the required tank capacity of the variable propellant case (solid curve) is less than that of the present design and it is assumed that the tanks are simply off-loaded rather than further decreasing their size. This is observed in the lower set of curves which represent the propulsion system inert (residual) mass corresponding to the delivered total orbiter burnout mass. The vertical distance between the two sets of curves is the mass available for all the spacecraft subsystems exclusive of propulsion.

From a short review of Mariner outer planet spacecraft design studies (including

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the JPL MJO Study for SAG, July 29, 1971) it would appear that 600 kg is a reasonable lower limit for spacecraft subsystems mass (excluding propulsion). This value should include at least 60 kg of science. Examining Figure 1, it is observed that this minimum value corresponds to a total burnout mass of 724 kg, or a maximum orbit periapse radius of 2 R_J, which in terms of orbit selection flexibility represents very little performance depth. Note, however, that this performance point is within the capability of the present MM '71 retro system, i.e. expanded tankage would not be necessary.

Proceeding to Figure 2, a similar set of orbiter mass curves are presented as a function of orbit periapse radius for Shuttle/Centaur/BII launched missions. The obvious difference is, of course, the increased injected payload capability of 1815 kg compared to 1165 kg with the Titan launch vehicle. With the present two-tank MM '71 design, however, the maximum periapse radius is still 2 R_J for a minimum orbiter mass of 724 kg (600 kg exclusive of propulsion). In other words, none of the Shuttle's improved performance can be realized unless the retro propellant capacity is increased. If, on the other hand, propellant capacity is increased to match Shuttle capability, then periapse radii up to 6.75 R_J are possible before spacecraft subsystems mass is reduced to 600 kg. At this point about 2.25 times as much propellant as MM '71 would have to be carried. A reasonable expanded design point might be double the capacity of the MM '71 propulsion subsystem in which case two more tanks of similar design would be added to the present configuration (assuming thermal control would still be possible). From Figure 2, a four-tank MM '71 propulsion design would provide orbit periapse flexibility up to 4.5 R_J without off-loading retro propellant. The minimum orbiter burnout mass would be about 920 kg, 730 kg of which could be allocated to spacecraft subsystems (including science and science support).

This quick-look at MJO missions with the earth-storable MM '71 propulsion subsystem provides two useful conclusions. First, the combination of Titan IIIE/Centaur/BII and earth-storable propulsion has marginal capability for MJO missions. Little additional performance depth could be gained by increasing orbit period, and other launch opportunities (specifically 1980 and 1983) will only further decrease performance. Second, the combination of Shuttle/Centaur/BII and earth-storable propulsion does provide acceptable MJO mission performance, but almost certainly requires redesign of the MM '71 propulsion subsystem. Specifically, doubling of the propellant capacity is indicated. Although the design feasibility of this modification is unclear, intuitively it doesn't appear unreasonable.

J. C. Niehoff

JCN/sn

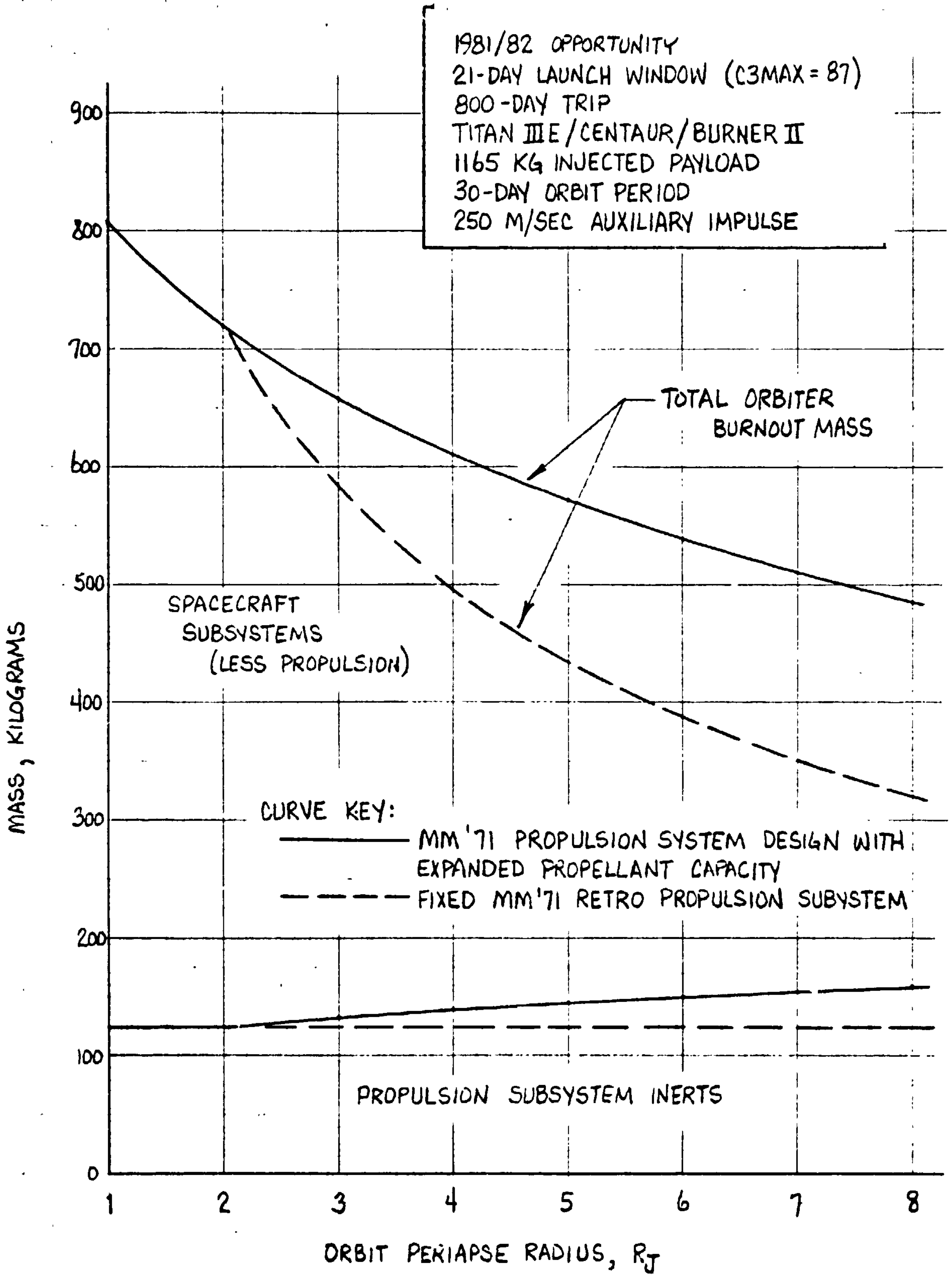


FIGURE 1. 1981/82 JUPITER ORBITER CAPABILITY WITH MM'71 PROPULSION (TITAN IIIE/CENTAUR/BII)

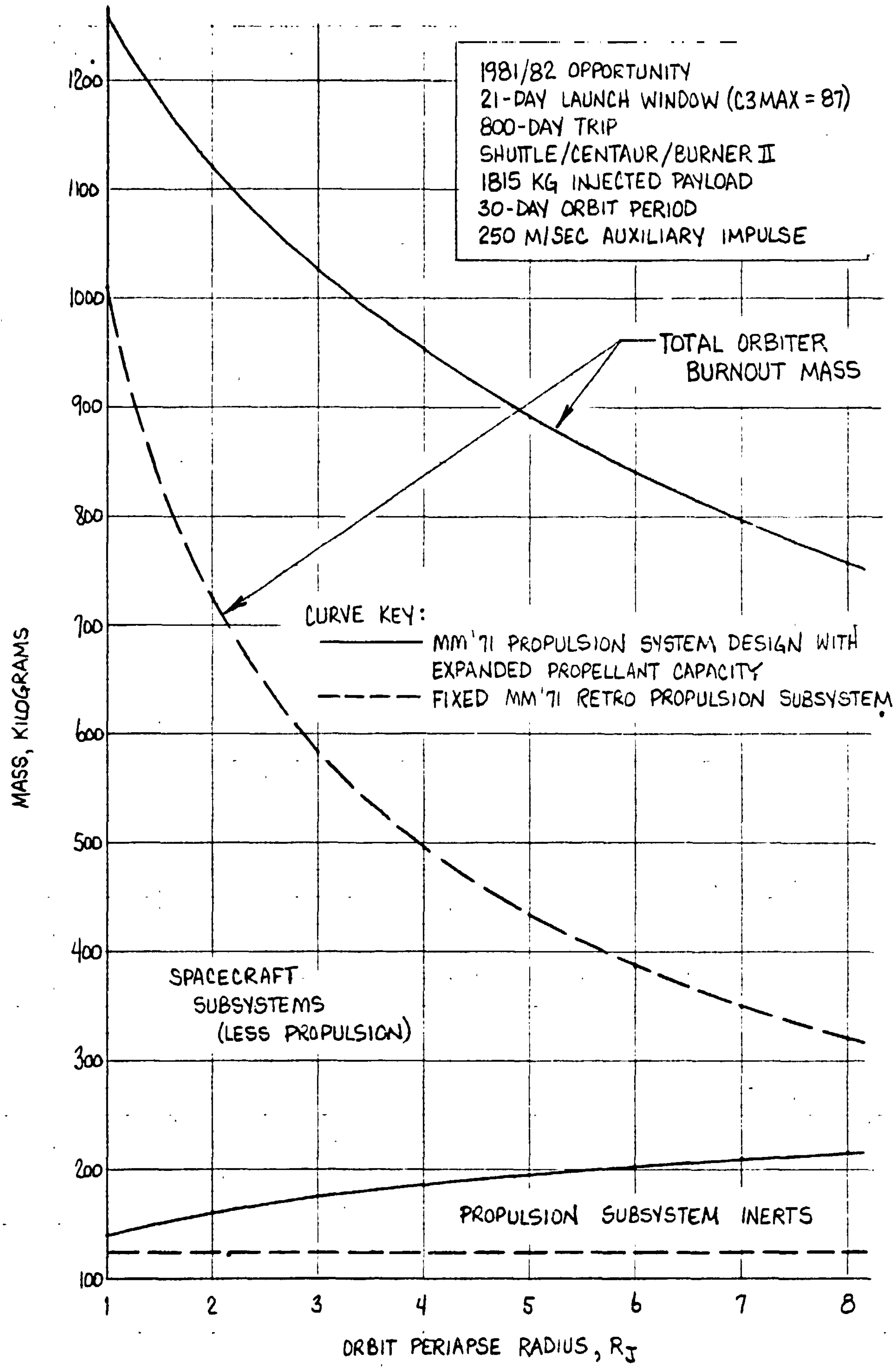


FIGURE 2. 1981/82 JUPITER ORBITER CAPABILITY (SHUTTLE/CENTAUR/BURNER II)

1983 VENUS AND 1986 URANUS/NEPTUNE SEP MISSIONS

Science Applications, Incorporated
Chicago O'Hare Aerospace Office Center
4825 North Scott Street, Suite 67, Schiller Park, Illinois 60176 (312) 678-4793

May 1, 1973

Mr. C. H. Guttman
Mail Stop PD-SA-P
Marshall Space Flight Center
National Aeronautics and Space Administration
Huntsville, Alabama 35812

Dear Chuck:

The attached tabular data describe our analysis of the 1983 Venus and 1986 Uranus/Neptune missions. Performance conditions assumed for the SEP stage are listed on each table, and the trajectory parameter notation is fairly standard. Three mass parameters are listed: (1) initial or injected mass, (2) propellant mass, and (3) net spacecraft mass at target approach. Note that the net mass includes all stage subsystems.

Data for the remaining reference missions will be sent to you as they are generated.

Sincerely,

Alan L. Friedlander

ALF/sn
Att.

cc: J. Gilbert, Rockwell International
D. Kerrisk, NASA Headquarters

(TABLE 1-A)

MIS: N: 1983 VENUS

NOMINAL POWER

$P_o = 21$ KW
$\alpha = 30$ KG/KW
$\Delta P/P_o = 0.02$

$I_{SP} = 3000$ SEC
$\eta = 0.66$
$K_P = 0.03$

$t_P = T_F$

T_F (DAYS)	T_L	V_{HL} (KM/SEC)	DLA (DEG)	M_o (KG)	M_P (KG)	APPROACH M_N (KG)	V_{HP} (KM/SEC)		
SHUTTLE	/TUG (I,R)								
70	7/12/83	4.95	3.5	2393	239	1517	6.708		
90	6/30/83	3.69	2.3	4442	315	3488	4.910		
120	6/13/83	2.35	1.5	5685	403	4640	3.282		
SHUTTLE	/TUG(I,R)	/KICK (APC)							
70	7/13/83	5.50	3.4	4400	248	3515	6.708		
90	7/1/83	3.90	2.0	6032	307	5086	4.910		
120	6/13/83	2.52	0.8	7185	402	6141	3.282		
SHUTTLE	/CENTAUR								
70	7/13/83	5.62	3.4	5250	230	4383	6.708		
90	7/1/83	3.99	1.8	7212	310	6263	4.910		
120	6/14/83	2.64	0.2	8955	500	7810	3.282		

$$G(R) = \begin{cases} 1.4382\,R^{-2} - 0.2235\,R^{-3} - 0.2147\,R^{-4} & \text{for } R \geq 0.68 \text{ AU} \\ 1.3952 & \text{for } R < 0.68 \text{ AU} \end{cases}$$

FORM 247 REV 12/55

(TABLE 1-B)

MISSION: 1983 VENUS
LOW POWER

$P_0 = 15$ kw
$\alpha = 40$ kg/kw
$\Delta P/P_0 = 0.02$

$I_{sp} = 3000$ SEC
$\eta = 0.66$
$K_p = 0.03$

$t_p = T_F$

T_F (DAYS)	T_L	V_{HL} (KM/SEC)	DLA (DEG)	M_0 (KG)	M_p (KG)	APPROACH M_N (KG)	V_{HP} (KM/SEC)		
SHUTTLE	TUG (I,R)								
70	7/12/83	5.16	3.5	2008	173	1230	6.708		
90	7/1/83	3.88	2.0	4174	230	3337	4.910		
120	6/13/83	2.57	0.4	5574	292	4673	3.282		
SHUTTLE	TUG (I,R)/KICK (APC)								
70	7/13/83	5.68	3.4	4218	174	3439	6.708		
90	7/1/83	4.05	1.7	5886	226	5053	4.910		
120	6/14/83	2.69	-0.4	7061	291	6161	3.282		
SHUTTLE	/CENTAUR								
70	7/13/83	5.76	3.4	5074	165	4304	6.708		
90	7/1/83	4.12	1.6	7058	217	6235	4.910		
120	6/14/83	2.78	-1.1	8810	387	7811	3.282		

$$G(R) = \begin{cases} 1.4382\,R^{-2} - 0.2235\,R^{-3} - 0.2147\,R^{-4} & \text{for } R \geq 0.68\,\text{AU} \\ 1.3952 & \text{for } R < 0.68\,\text{AU} \end{cases}$$

FORM 247 REV 12/55

MISS. J: 1986 URANUS/NEPTUNE

(TABLE 2)

$P_0 = 21$ KW
$\alpha = 30$ KG/KW
$\Delta P/P_0 = 0.02$

$I_{SP} = 3000$ SEC
$\eta = 0.61$
$K_P = 0.03$

$t_P = 350^d$

$T_{F,U}$ (YEARS)	$T_{F,N}$ (YEARS)	T_L	V_{HL} (KM/SEC)	DLA (DEG)	V_{HU} (KM/SEC)	R_{PU} (RAW)	V_{HN} (KM/SEC)	M_0 (KG)	M_P (KG)	APPROACH M_N (KG)
SHUTTLE/	TUG (I,R)/	KICK (APC)								
4.32	6.97	1/19/86	9.32	−17.6	18.93	3.0	19.51	1340	357	342
4.86	7.93	"	8.97	−16.4	16.36	4.3	16.70	1532	366	525
5.10	8.37	"	8.84	−15.8	15.39	5.1	15.58	1608	367	600
5.79	9.66	"	8.52	−14.5	13.06	7.9	13.00	1806	378	787
SHUTTLE	CENTAUR									
4.32	6.97	1/19/86	9.76	−17.9	18.91	3.0	19.51	1419	360	418
4.86	7.93	"	9.45	−16.9	16.35	4.3	16.70	1635	368	626
5.10	8.37	"	9.33	−16.4	15.38	5.1	15.58	1720	363	716
5.79	9.66	"	9.02	−15.2	13.05	7.9	13.00	1945	362	942

FORM 247 REV 12/55

1989 VENUS AND 1981/82 ENCKE RENDEZVOUS
SEP MISSIONS

Science Applications, Incorporated
Chicago O'Hare Aerospace Office Center
4825 North Scott Street, Suite 67, Schiller Park, Illinois 60176 (312) 678-4793

May 3, 1973

Mr. C. H. Guttman
Mail Stop PD-SA-P
Marshall Space Flight Center
National Aeronautics and Space Administration
Huntsville, Alabama 35812

Dear Chuck:

The attached tabular data describe our analysis of the 1989 Venus and 1981/82 Encke Rendezvous missions. Note that a low power (15 kw) option for the Venus mission is not given since the previous data submission showed no significant advantage for this option. Also note that the Shuttle/Tug without kick stage has inadequate performance for the shorter (750 - 800 day) missions to Comet Encke, hence, the data is not shown.

Sincerely,

Alan

Alan L. Friedlander

ALF/sn
Enc.

cc: Gilbert
Kenick

(TABLE 3)

MISSION : 1989 VENUS

$P_0 = 21$ kW
$\alpha = 30$ KG/kW
$\Delta P/P_0 = 0.02$

$I_{sp} = 3000$ SEC
$\eta = 0.66$
$K_p = 0.03$

$t_p = T_F$

T_F (DAYS)	T_L	V_{HL} (KM/SEC)	DLA (DEG)	M_0 (KG)	M_P (KG)	APPROACH M_N (KG)	V_{HP} (KM/SEC)		
SHUTTLE/TUG (I, R)									
70	12/7/89	4.77	-9.1	2717	254	1825	5.845		
90	11/25/89	3.60	-5.7	4563	316	3608	4.171		
120	11/10/89	2.45	-4.8	5640	415	4583	2.635		
SHUTTLE/TUG(I,R)/KICK(APC)									
70	12/7/89	5.29	-8.2	4615	246	3732	5.845		
90	11/26/89	3.83	-4.6	6100	311	5150	4.171		
120	11/11/89	2.73	-2.0	7032	416	5974	2.635		
SHUTTLE/CENTAUR									
70	12/7/89	5.40	-8.0	5523	250	4635	5.845		
90	11/26/89	3.94	-4.0	7271	312	6320	4.171		
120	11/12/89	2.95	0.5	8585	452	7489	2.635		

$$G(R) = \begin{cases} 1.4382\,R^{-2} - 0.2235\,R^{-3} - 0.2147\,R^{-4} & \text{for } R \geq 0.68 \text{ AU} \\ 1.3952 & \text{for } R < 0.68 \text{ AU} \end{cases}$$

FORM 247 REV 12/55

MISSION. 1981/82. ENCKE RENDEZVOUS

$P_0 = 21$ kw
$\alpha = 30$ KG/KW
$\Delta P/P_0 = 0.02$

(TABLE 4-A)

$I_{sp} = 3000$ SEC
$\eta = 0.63$
$K_p = 0.03$

t_p = PROPULSION TIME

T_F (DAYS)	$T_{ARRIVAL}$	T_L	t_p (DAYS)	V_{HL} (KM/SEC)	DLA (DEG)	M_0 (KG)	M_P (KG)	APPROACH M_N (KG)	
SHUTTLE/CENTAUR									
1100	T_p-30^d	2/21/81	1100	7.72	-45.1	3031	820	1556	
			1000	8.23	-47.5	2583	596	1339	
			900	8.73	-49.9	2175	440	1092	
1080	T_p-50^d	2/21/81	1080	7.93	-44.7	2842	762	1427	
			1000	8.34	-46.4	2491	588	1255	
			900	8.76	-48.1	2151	438	1070	
800	T_p-50^d	11/28/81	800	8.43	-27.8	2417	1007	750	
	T_p-30^d	12/18/81	800	8.15	-31.6	2651	1019	971	
750	T_p-50^d	1/17/82	750	8.78	-35.8	2135	881	598	
	T_p-30^d	2/6/82	750	8.42	-42.3	2425	892	876	

FORM 247 REV 12/59

MISSION 1981/82 ENCKE RENDEZVOUS

$P_0 = 21$ kw
$\alpha = 30$ kg/kw
$\Delta P/P_0 = 0.02$

(TABLE 4-8)
$I_{SP} = 3000$ SEC
$\eta = 0.63$
$K_P = 0.03$

t_P = PROPULSION TIME

T_F (DAYS)	$T_{ARRIVAL}$	T_L	t_P (DAYS)	V_{HL} (KM/SEC)	DLA (DEG)	M_0 (KG)	M_P (KG)	APPROACH M_N (KG)	
SHUTTLE	/ TUG (I, R)	/ KICK (APC)							
1100	$T_P - 30^d$	2/21/81	1100	6.99	-41.4	2971	822	1494	
			1000	7.48	-43.9	2559	624	1286	
			900	8.05	-46.6	2126	444	1039	
1080	$T_P - 50^d$	2/21/81	1080	7.19	-41.7	2798	772	1373	
			1000	7.58	-43.3	2479	609	1222	
			900	8.09	-45.4	2097	440	1014	
800	$T_P - 50^d$	11/28/81	800	7.70	-28.1	2386	1017	709	
	$T_P - 30^d$	12/18/81	800	7.39	-32.0	2632	1043	928	
750	$T_P - 50^d$	1/17/82	750	8.08	-36.2	2105	880	569	
	$T_P - 30^d$	2/6/82	750	7.70	-42.3	2386	893	836	

FORM 247 REV 12/55

MISSION 1981 ENCKE RENDEZVOUS

(TABLE 4-C)

$P_0 = 21$ kW
$\alpha = 30$ kg/kW
$\Delta P/P_0 = 0.02$

$I_{sp} = 3000$ SEC
$\eta = 0.63$
$K_p = 0.03$

t_p = PROPULSION TIME

T_F (DAYS)	$T_{ARRIVAL}$	T_L	t_p (DAYS)	V_{HL} (KM/SEC)	DLA (DEG)	M_0 (KG)	M_P (KG)	APPROACH M_N (KG)
SHUTTLE/TUG (I, R)								
1100	$T_p - 30^d$	2/21/81	1100	4.82	-31.4	2627	825	1147
			1000	4.99	-32.1	2320	682	988
			900	5.17	-32.8	1990	538	806

1989 SATURN AND 1989 ASTEROID (METIS) RENDEZVOUS SEP MISSIONS

Science Applications, Incorporated
Chicago O'Hare Aerospace Office Center
4825 North Scott Street, Suite 67, Schiller Park, Illinois 60176 (312) 678-4793

May 8, 1973

Mr. C. H. Guttman
Mail Stop PD-SA-P
Marshall Space Flight Center
National Aeronautics and Space Administration
Huntsville, Alabama 35812

Dear Chuck:

The attached tabular data describe our analysis of the 1989 Saturn and 1989 Metis (asteroid rendezvous) missions. The choice of asteroid was made after discussion with Dr. C. Chapman (Planetary Science Institute - SAI). Metis appears to be quite interesting from a scientific standpoint in that ground-based observations show it to have a reddish color and high albedo. Surface characteristics tend toward meteoritic material (rocky), and since it is fairly large it is likely to be differentiated. The orbital elements of Metis are:

$$a = 2.386 \text{ AU}$$
$$e = 0.123$$
$$i = 5.^{\circ}6$$
$$\Omega = 69.^{\circ}6$$
$$\bar{\omega} = 72.^{\circ}3$$
$$T_p = 27 \text{ Aug. } 1990$$

Sincerely,

Alan

Alan L. Friedlander

ALF/sn
Att.

cc: J. Gilbert, Rockwell International
D. Kerrisk, NASA Headquarters

MISSION: I' 9 SATURN

(TABLE 5)

$P_0 = 21$ kw, $\alpha = 30$ kg/kw, $\Delta P/P_0 = 0.02$

$I_{sp} = 3000$ SEC, $\eta = 0.61$, $k_p = 0.03$

$t_p = 350^d$

T_F (DAYS)	T_L	V_{HL} (KM/SEC)	DLA (DEG)	M_0 (KG)	M_P (KG)	APPROACH M_N (KG)	V_{HP} (KM/SEC)		
SHUTTLE/TUG(I R)/KICK (APC)									
1400	2/13/89	7.55	-20.5	2503	310	1554	8.60		
1600	2/13/89	7.39	-19.9	2632	313	1680	7.11		
1800	2/14/89	7.33	-19.5	2681	308	1734	6.13		
2000	2/15/89	7.25	-19.3	2748	322	1786	5.55		
SHUTTLE/CENTAUR									
1400	2/16/89	7.96	-21.0	2816	306	1871	8.57		
1600	2/15/89	7.80	-20.4	2962	307	2016	7.10		
1800	2/16/89	7.74	-20.0	3017	299	2079	6.13		
2000	2/18/89	7.67	-19.8	3081	310	2132	5.55		

FORM 247 REV 12/59

(TABLE 1-A)

MISSION: '89 METIS RENDEZVOUS

$\alpha = 30$ kg/kW
$I_{sp} = 3000$ SEC
$\eta = 0.63$

$k_p = 0.03$
$\Delta P/P_0 = 0.031$ (650w @ 21kw)
$t_p = T_F$

- NORMALIZED PAYLOAD DATA

T_F (DAYS)	T_L 244-	V_{HL} (KM/SEC)	DLA (DEG)		P_0/M_0 kW/1000 KG	M_N/M_0			
450	7638	4.0	-38.1		13.36	0.292			
	7642	4.5	-40.0		13.20	0.305			
	7645	5.0	-41.9		13.07	0.316			
	7652	6.0	-44.9		12.96	0.329			
	7659	7.0	-46.1		12.91	0.337			
550	7624	4.0	-37.6		9.03	0.488			
	7630	4.5	-41.2		9.02	0.492			
	7635	5.0	-44.3		9.03	0.495			
	7644	6.0	-48.0		9.11	0.497			
	7652	7.0	-49.2		9.16	0.499			
650	7615	4.0	-38.1		7.53	0.545			
	7623	4.5	-42.7		7.58	0.546			
	7630	5.0	-45.8		7.70	0.543			
	7641	6.0	-48.9		7.86	0.540			
	7650	7.0	-49.8		8.01	0.536			

MISSION 1989 METIS RENDEZVOUS

(TABLE 6-B)

$P_0 = 21k$ $\quad I_{sp} = 3000$ SEC

$\alpha = 30$ KG/KW $\quad \eta = 0.63$

$\Delta P/P_0 = 0.031$ $\quad k_p = 0.03$ $\quad t_p = T_F$

T_F (DAYS)	T_L	V_{HL} (KM/SEC)	DLA (DEG)	M_0 (KG)	M_P (KG)	APPROACH M_N (KG)		
ALL 3	LAUNCH	VEHICLES	(OFF-LOADED	CONDITIONS)				
450	4/28/89	5.0	-41.9	1607	455	508		
550	4/13/89	4.5	-41.2	2328	537	1145		
650	3/29/89	4.0	-38.1	2789	620	1520		

1987 MERCURY SEP MISSION

Science Applications, Incorporated
Chicago O'Hare Aerospace Office Center
4825 North Scott Street, Suite 67, Schiller Park, Illinois 60176 (312) 678-4793

May 23, 1973

Mr. C. H. Guttman
Mail Stop PD-SA-P
Marshall Space Flight Center
National Aeronautics and Space Administration
Huntsville, Alabama 35812

Dear Chuck:

This is the final submission of trajectory data in regard to our support of Rockwell International in their continuing study of SEP stage performance. The tabular data below is for a 450-day mission to Mercury (as requested by Ed Dazzo) using a launch velocity compatible with the interim, reusable Tug capability. The limited amount of data is due to the difficulty and expense of obtaining converged Mercury trajectories which was experienced; the launch dates listed are "near optimum".

$T_F = 450^d$ $\alpha = 30$ kg/kw $I_{sp} = 3000$ sec

$P/P_o = 1.0$ $k_p = 0.03$ $\eta = 0.67$

1. Propulsion time = flight time

Launch Date	V_{HL} (km/sec)	V_{HP} (km/sec)	P_o/M_o (kw/1000 kg)	Approach M_N/M_o	at P_o = 21 kw M_o (kg)	M_N (kg)
5/14/87	4.0	2.0	6.18	0.462	3398	1570
5/29/87	4.0	2.0	6.32	0.449	3323	1492
5/29/87	4.5	2.0	6.29	0.451	3339	1506
6/3/87	4.0	0	7.06	0.381	2975	1133

Cont..

2. Reduced Propulsion Time (400 & 350 days)

5/14/87	4.0	2.0	6.76	0.455	3107	1413
5/14/87	4.0	2.0	7.52	0.442	2793	1234

Sincerely,

Alan

Alan L. Friedlander

ALF/sn

cc: J. Gilbert, Rockwell International
D. Kerrisk, NASA Headquarters

SPACE SHUTTLE AND PLANETARY MISSIONS

(Copy of Full Report Available from S. Grivas, Code SL, NASA HQ.)

SPACE SHUTTLE
AND
PLANETARY MISSIONS

MAY 1973

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

THE SPACE SHUTTLE AND PLANETARY MISSIONS
MAY 1973

INTRODUCTION

The purpose of this paper is to review and discuss the application of the Space Shuttle system to planetary missions, particularly during its introductory years of service, 1980-85. It is the intent here to relate anticipated planetary mission requirements with candidate Shuttle-based escape stage capabilities. In addition, several specific mission point designs are detailed on the basis of a Shuttle/Centaur launch system. The reader is cautioned that the Shuttle escape stage data presented is preliminary in nature and still under study.

The paper is organized into several sections. The first section presents the current mission model and the rationale related to these future plans.

Section 2 includes a brief description of the Shuttle and its operations for planetary missions. Several escape-stage alternatives are presented including the Centaur, the recoverable and expendable Tugs. An escape-stage capture evaluation is presented for nine different planet, comet, and asteroid missions assuming a 20 KW solar electric propulsion (SEP) stage is available as needed.

Section 3 is comprised of three mission descriptions assuming a Shuttle/Centaur launch system is used for these missions. The missions considered are: (a) 1980 Pioneer Saturn/Uranus Entry Probes, (b) 1981 Encke SEP Rendezvous, and (c) 1981/82 Mariner Jupiter Orbiters. Benefits of using the Shuttle/Centaur rather than the Titan IIID/Centaur are discussed.

CONTENTS

	Page
INTRODUCTION	iii
1. RATIONALE FOR PLANETARY EXPLORATION PROGRAM	1
1.1 INNER PLANETS	1
1.2 OUTER PLANETS	4
1.3 COMETS AND ASTEROIDS	5
2. SHUTTLE PLANETARY APPLICATIONS	6
2.1 SHUTTLE PLANETARY MISSION OPERATIONS	6
2.2 SHUTTLE-BASED ESCAPE STAGE PERFORMANCE	8
3. MISSION EXAMPLES WITH SHUTTLE/CENTAUR	14
3.1 1980 PIONEER SATURN/URANUS ENTRY PROBES	15
3.2 1981 ENCKE RENDEZVOUS	22
3.3 1981/82 MARINER JUPITER ORBITER	29
REFERENCES	35

TABLES

Table		Page
1	Shuttle-Based Propulsion Stage Candidates	10
2	Planetary Mission Point Designs	11
3	Pioneer/Probe Saturn-Uranus Objectives and Implementation	16

TABLES (Continued)

Table		Page
4	Summary of Pioneer/Probe 1980 S/U Mission Parameters . . .	18
5	Encke Rendezvous Objectives and Implementation	24
6	Summary of 1981 Encke Rendezvous Mission Parameters	26
7	Mariner Jupiter Orbiter Objectives and Implementation	30
8	Summary of 1981/82 Mariner Jupiter Orbiter Mission Parameters	32

PIONEER SATURN AND URANUS
ENTRY PROBE MISSION DATES

Science Applications, Incorporated
Chicago O'Hare Aerospace Office Center
4825 North Scott Street, Suite 67, Schiller Park, Illinois 60176 (312) 678-4793

May 21, 1973

Dr. James A. Van Allen, Head
Dept. of Physics and Astronomy
University of Iowa
Iowa City, Iowa 52240

Subject: Pioneer Saturn and Uranus Entry Probe Mission Dates

Dear Jim:

Late in the OPSAC outer planet entry probe discussions last Friday, John Wolfe suggested that the "Niehoff - Cameron Plan 1" be altered from the Pioneer/Probe set

1980 PJU, 1980 PS, and 1981 PSU

to the following set with more targetting flexibility:

1980 PJU, 1981 PS, and 1982 PSU.

I promised to investigate the feasibility of this set, particularly with regard to the third mission's (1982 PSU) targetting options prior to Saturn encounter.

The results of a quick-look at the revised mission set are summarized in Table 1. Basically, the advantage of targetting flexibility is preserved on the third mission, now the 1982 PSU mission. As can be seen from the mission schedule in the table, all dates have slipped about a year from the earlier plan I presented last Thursday.

Specific mission parameters are presented at the bottom of Table 1. The Jupiter swingby radius for the 1980 PJU, 12.3 R_J no longer represents a potential radiation hazard. The trip time to Uranus is just under 5 years, about 95 days longer than the 1979 PJU. This puts the Uranus

Cont..

entry closer to the second Saturn encounter, 4 months before rather than 8 months. (Note that the Saturn encounter data of the 1981 PSU mission in my handout to the OPSAC is incorrectly shown at 12/5/84; it should be 3/15/85.) Hence, although Titan and Uranus targetting would still be possible following the 11/30/85 Uranus entry, it would probably not be possible to recover a second Saturn entry at this late date. The remaining parameters are quite similar to the earlier data I presented.

I trust the enclosed material is sufficient for the final OPSAC report you're drafting. If you have any questions, please call me, (312/678-4793).

Sincerely yours,

John

John C. Niehoff

JCN/sn

cc: D. Herman/SL
J. Long/JPL
J. Wolfe/ARC

TABLE 1

REVISED PIONEER SATURN AND URANUS ENTRY PROBE STRATEGY

- **KEY MISSION DATES**

CY	80	81	82	83	84	85	86	87	88	89	90	91
o 1980 JUPITER/URANUS SWINGBY	L 12/2		J 3/8			U 11/30						
o 1981 SATURN DIRECT		L 12/6				S 3/15						
o 1982 SATURN/URANUS SWINGBY			L 12/17				S 3/30				U 5/13	

- **KEY MISSION PARAMETERS**

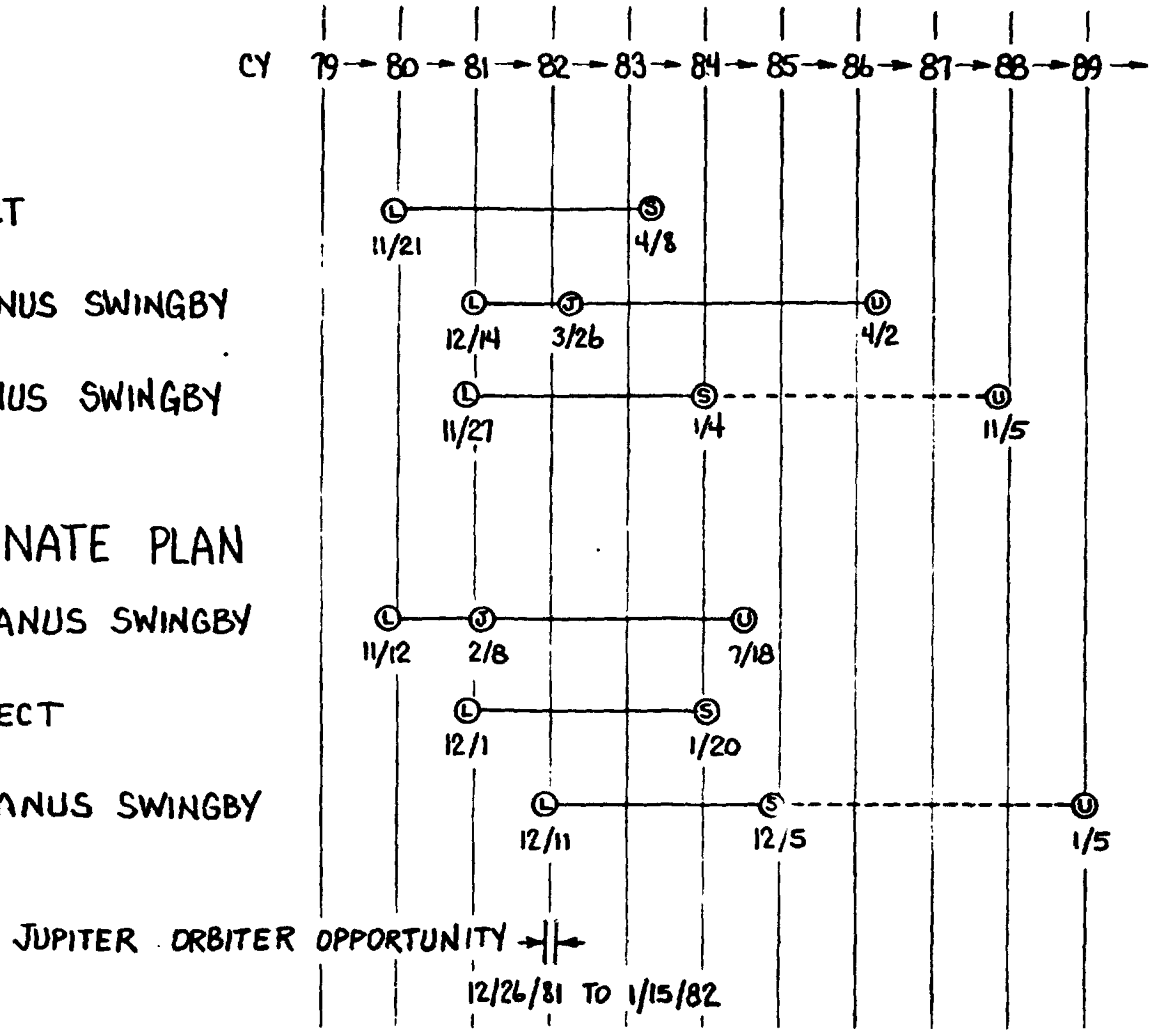
	'80 J/U	'81 S	'82 S/U
o LAUNCH ENERGY, C3, $(km/sec)^2$	142*	142*	142*
o LAUNCH WINDOW, DAYS	15	16	18
o JUPITER SWINGBY RADIUS, R_J	12.3	-	-
o SATURN SWINGBY RADIUS, R_S	-	-	9.4
o TRIP TIME TO JUPITER, YEARS	1.24	-	-
o TRIP TIME TO SATURN, YEARS	-	3.25	3.26
o TRIP TIME TO URANUS, YEARS	4.97	-	7.37

* NOTE: TITAN IIIE/CENTAUR/TE 364-4 CAPABILITY @ 142 = 475 KG

JCN/SAI

PIONEER SATURN AND URANUS ENTRY PROBE MISSION STRATEGIES

- PRESENT PLAN
  - o 1979 SATURN DIRECT
  - o 1980 JUPITER/URANUS SWINGBY
  - o 1980 SATURN/URANUS SWINGBY
- SUGGESTED ALTERNATE PLAN
  - o 1979 JUPITER/URANUS SWINGBY
  - o 1980 SATURN DIRECT
  - o 1981 SATURN/URANUS SWINGBY



OUTER PLANET PIONEER/PROBE MISSION CHARACTERISTICS

MASS SUMMARY*

● 10-BAR PROBE		113 KG+
SCIENCE	9	
BUS	64	
HEAT SHIELD	40	
● PIONEER		357
SCIENCE	28	
BUS	246	
PROBE ADAPTER	30	
PROPELLANT	53	
● TOTAL		470 KG
● TITAN IIID/CENTAUR/TE 364-4 CAPABILITY @ $C3 = 142\ (KM/SEC)^2$		475 KG

* BASED ON MacDAC AND TRW PHASE-B MIDTERM BRIEFINGS
+ INCLUDES 15% GROWTH MARGIN

MISSIONS SUMMARY

	C3	LW	RJ	RS	TFJ	TFS	TFU
● PRESENT PLAN							
○ 1979 SATURN FB	142	15[d]	—	—	—	3.4[γ]	—
○ 1980 J/U SWB	142	12*	14.5	—	1.3[γ]	—	5.3[γ]
○ 1980 S/U SWB	142	11*	—	2.7	—	3.1	6.9
● ALTERNATE PLAN							
○ 1979 J/U SWB	142	13	3.3	—	1.2	—	4.7
○ 1980 SATURN FB	142	11	—	—	—	3.1	—
○ 1981 S/U SWB	142	16	—	5.0	—	3.3	7.1

* SERIAL LAUNCH WINDOWS

SUMMARY OF PIONEER/PROBE ALTERNATE PLAN

- DISADVANTAGES
  - O REQUIRED CLOSE JUPITER FLYBY ($3.3\ R_J$) OF '79 PJU MISSION STILL QUESTIONABLE
  - O SECOND URANUS ENTRY (IF CHOSEN) NOT UNTIL JAN/89; THIS COULD BE DESIRABLE IN THAT THE URANUS NORTH POLE HAS ROTATED ~20° SINCE FIRST URANUS ENTRY (7/84)
  - O THERE MAY BE LAUNCH SCHEDULE DIFFICULTIES WITH PLANNED '79 MJU AND '81/82 MJO MISSIONS

- ADVANTAGES
  - O THE THIRD MISSION IS NOT COMMITTED TO AN ENTRY TARGET UNTIL 30-45 DAYS AFTER FIRST TWO ENTRIES HAVE BEEN COMPLETED
  - O THE FIRST URANUS ENTRY OCCURS ALMOST TWO YEARS SOONER (7/84), ABOUT SIX MONTHS AFTER FIRST SATURN ENTRY (1/84)
  - O THE SATURN SWINGBY FOR THE '81 PSU MISSION IS SAFER; SWINGBY IS AT $5R_S$ VERSUS $2.7R_S$ FOR '80 PSU MISSION

- COMMENTS
  - O RETARGETING '81 S/U MISSION ($5R_S \rightarrow 0.9\ R_S$) FOR SATURN ENTRY (IF CHOSEN) AT SEVEN MONTHS BEFORE ENCOUNTER IS WITHIN PIONEER CAPABILITY (25 M/SEC REQUIRED)
  - O IF SHUTTLE/CENTAUR/TE 364-4 USED FOR '81 PSU, IT IS EASILY LAUNCHED BEFORE '81/82 MJO LAUNCH WINDOW OPENS
  - O LAUNCH WINDOWS ARE COMPARABLE IN EITHER PLAN; FLIGHT TIMES ARE COMPARABLE OR SHORTER IN ALTERNATE PLAN

COMET KOHOUTEK FLYBY MISSION PARAMETERS

Science Applications, Incorporated
Chicago O'Hare Aerospace Office Center
4825 North Scott Street, Suite 67, Schiller Park, Illinois 60176 (312) 678-4793

May 29, 1973

Mr. Daniel Kerrisk
Advanced Programs and Technology
Planetary Programs Division, Code SL
National Aeronautics and Space Administration
Washington, D. C. 20546

Subject: Comet Kohoutek Flyby Mission Parameters

Dear Dan:

Enclosed in Table 1 are optimum flyby trajectory parameters for Comet Kohoutek as a function of launch date during the next six months. Encounter dates occur early next year as the comet passes through its descending node after perihelion. Note that the trajectories presented go out as far as 1.8 AU prior to (or at) encounter with spacecraft-earth communication distance at flyby reaching 2 AU. Launches much after Labor Day are probably unrealistic due to high launch C3 requirements. A plot of injected payload performance versus launch date is presented in Figure 1 for three launch vehicles: 1) the Scout E, 2) the Delta 2914, and 3) the Atlas D/Centaur/TE 364-4. The Scout E is obviously too small a launch vehicle for sensible spacecraft payloads. A navigatible spacecraft capable of communicating with earth from a distance of up to 2 AU probably weighs at least 200 kg including science instruments. For 200 kg payload the Delta 2914 can meet Kohoutek flyby mission launch requirements until 11 August 1973; the Atlas D/Centaur/TE 364-4 can do so until 8 September. I trust the enclosed information is sufficient for present purposes. Please call me if you wish to pursue this matter further.

Sincerely yours,

John

John C. Niehoff

JCN/sn

TABLE 1

MINIMUM ENERGY TRANSFER CHARACTERISTICS FOR COMET KOHOUTEK FLYBY MISSIONS

LAUNCH DATE	TIME OF FLIGHT (DAYS)	ARRIVAL DATE	LAUNCH PARAMETERS		MAXIMUM S/C SOLAR DISTANCE (AU)	FLYBY PARAMETERS	
			C3 (KM^2/SEC^2)	DLA (DEG)		VHP (KM/SEC)	R_c* (AU)
5/23/73	265	2/16/74	5.5	-3.6	1.36	37.5	1.35
6/16/73	245	2/16/74	8.5	-29.7	1.37	37.5	1.35
7/06/73	233	2/24/74	13.3	-22.0	1.50	34.4	1.56
7/26/73	220	3/03/74	23.4	-14.8	1.64	31.5	1.77
8/15/73	209	3/12/74	45.6	-9.5	1.79	28.9	2.01
9/04/73	151	3/14/74	91.9	-5.6	1.82	28.3	2.06
9/24/73	161	3/04/74	179.5	-2.9	1.66	31.5	1.80
10/14/73	126	2/17/74	317.7	-0.9	1.41	38.7	1.38
11/03/73	93	2/04/74	512.9	0.5	1.23	48.5	1.06
11/23/73	67	1/29/74	815.5	2.1	1.11	55.2	0.94

* R_c: ENCOUNTER COMMUNICATION DISTANCE TO EARTH

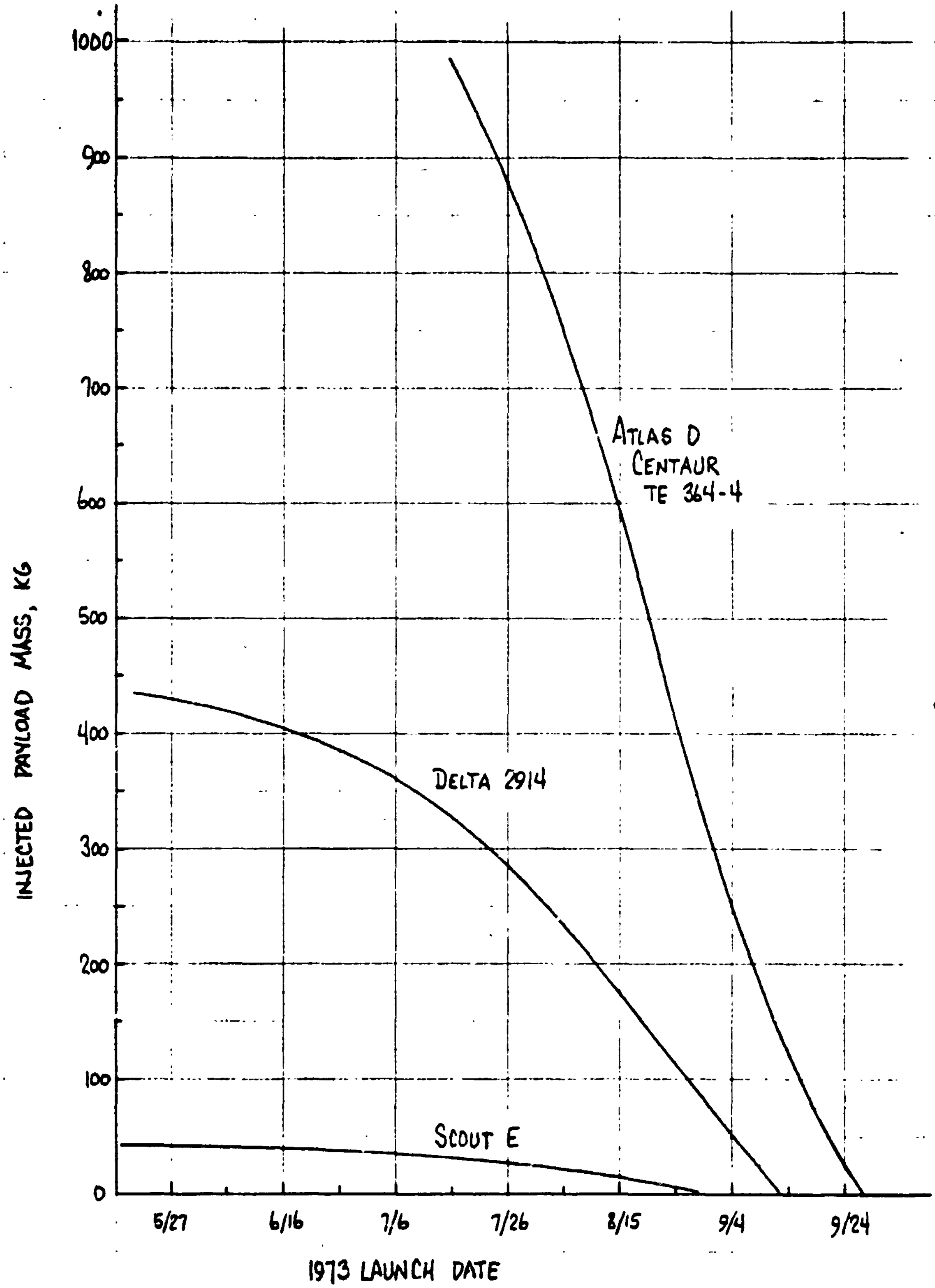
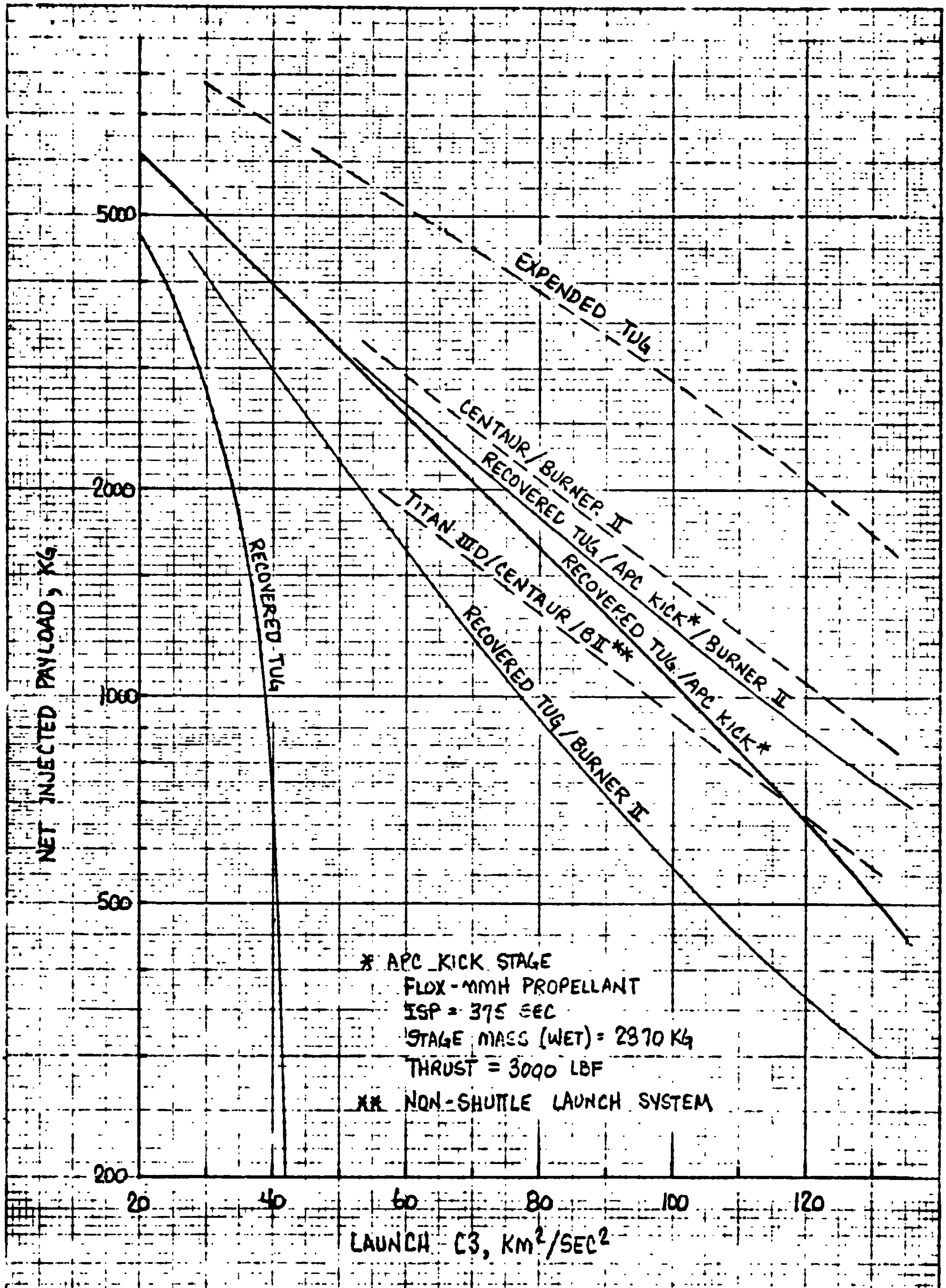


FIGURE 1. LAUNCH VEHICLE PERFORMANCE FOR COMET KOHOUTEK FLYBY MISSIONS

JCN/SAI

RECOVERED TUG EARTH ESCAPE PERFORMANCE
FOR PLANETARY MISSIONS



RECOVERED TUG EARTH ESCAPE PERFORMANCE WITH KICK STAGES

TITAN ATMOSPHERE WORKSHOP

Titan Atmosphere Workshop (July 25-27, 1973)

The workshop was convened at Ames Research Center under the chairmanship of D. M. Hunten. At the request of NASA Headquarters, the purpose was to define, as far as now possible, the atmosphere of Titan for use in the planning of future missions to that body. Titan's prominence is so recent that all the active workers could easily meet in a small room. More than half these people were actually present, and a good coverage of the appropriate disciplines was obtained.

Titan offers a unique opportunity in solar system exploration. It is the smallest known body with an atmosphere. In terms of spacecraft entry dynamics, it has the most accessible atmosphere in the solar system. It has dark reddish clouds which many workers believe are composed of organic compounds, falling from the sky like manna from heaven. It has the highest ratio of methane to hydrogen of all known reducing atmospheres, making an environment in some respects like that of the primitive earth at the time of the origin of life. It probably has the only surface of all the bodies beyond Mars with atmospheres that entry spacecraft can reach. In terms of planetary rotation rate, Titan's atmospheric circulation may occupy ' 'nique niche between the dynamics of Venus and the earth. The surface temperature may be 150 - 200^{O}K or warmer, and one model suggests an ocean of liquid methane and ammonia. While at the present this is the merest speculation, the presence of life on Titan is by no means out of the question.

Nearly two of the three days of the workshop were devoted to review papers, more than half of which concerned, as yet, unpublished results of Titanian studies and observations. The whole of our present knowledge of Titan was found to be clearly inadequate for engineering purposes (specifically atmospheric modeling), but it was equally clear that a vast improvement is feasible with today's observational techniques. These include ultra-violet and infrared spectroscopy, infrared and microwave radiometry, and stellar occultations. Observational and modeling techniques that have been used to study the planets have just begun to be applied to Titan. Many important properties are accessible which will yield a considerable improvement in our knowledge of Titan in the near future. Half a dozen recommendations for immediate work, both at the telescope and in the laboratory were generated by the workshop participants.

It was recognized, however, that a thorough characterization of the environment of Titan -- and, in particular, studies of the tantalizing questions of organic chemistry and surface morphology -- must await spacecraft investigations at or near Titan. With respect to mission planning, it was concluded that although the Mariner Jupiter/Saturn flyby missions, presently planned for launch in 1976, do not appear essential to the preparation of an atmospheric probe mission to Titan, the inclusion of Titan flyby objectives on the MJS missions would be most useful. It was also the consensus of the participants that the present outer planets atmospheric probe mission plan does not have sufficient emphasis for Titan. In particular, the three-mission set of Pioneer-Entry Probe missions includes Titan as a possible target of opportunity after Saturn and Uranus. A five-mission set of Pioneer-Probe's, with two launches dedicated for Titan, seems more appropriate. Questions regarding relevant probe science and sterilization were also discussed.

A draft report of the Titan Workshop proceedings will be available mid-November. Final publication of the Proceedings as a NASA-SP document is planned.

As editor for the Workshop, there was considerable coordination required during the meeting to obtain preliminary copies and transcripts of all presentations and discussions. This was followed by a concerted effort to compile a final draft version of the proceedings of the Workshop within a matter of weeks after the meeting. The report finally appeared as a NASA Special Publication, SP 340.

INPUTS FOR ELECTRIC PROPULSION CONFERENCE

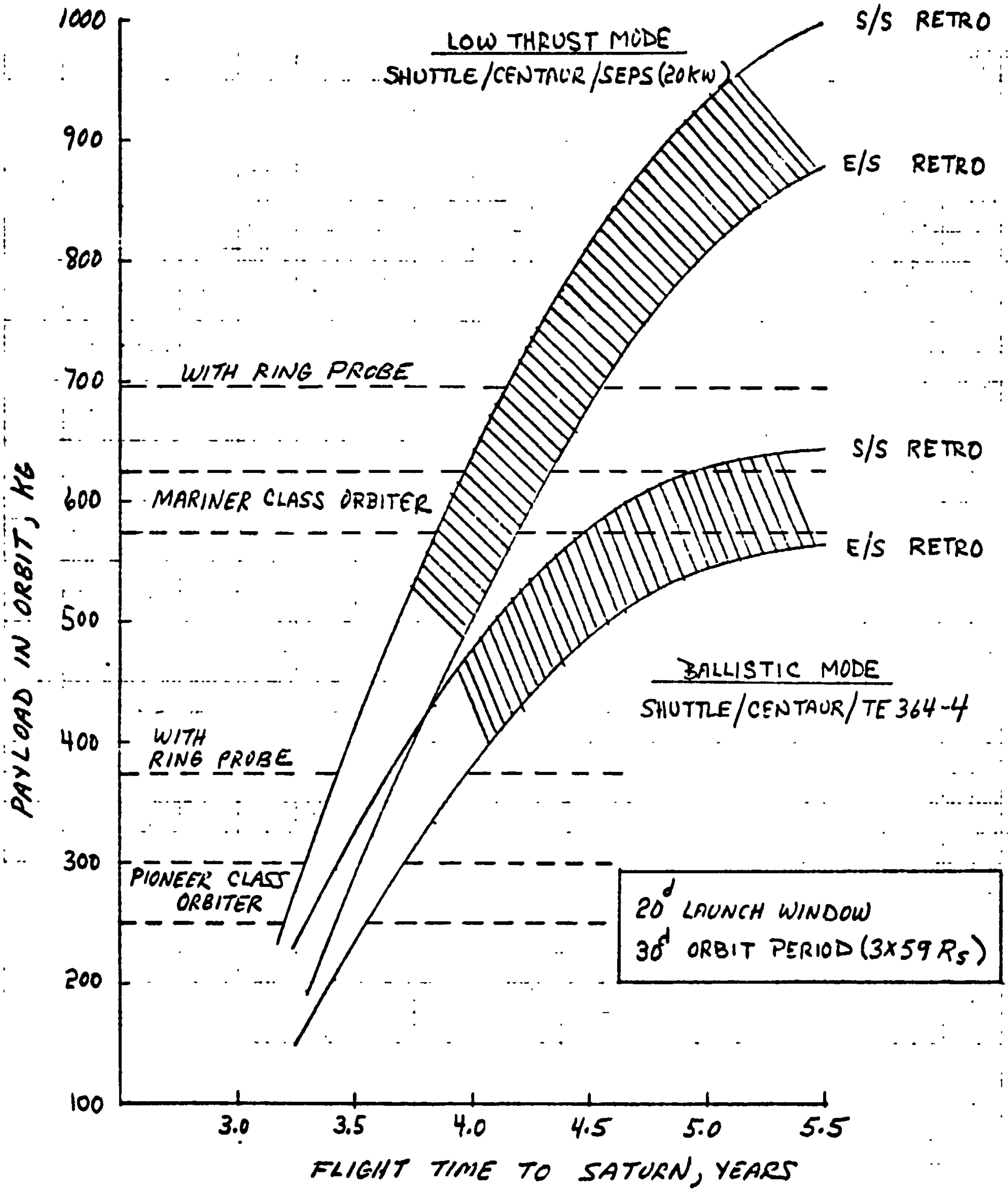
DRAFT ILLUSTRATIONS FOR

DAN HERMAN'S TALK

ELECTRIC PROPULSION CONFERENCE

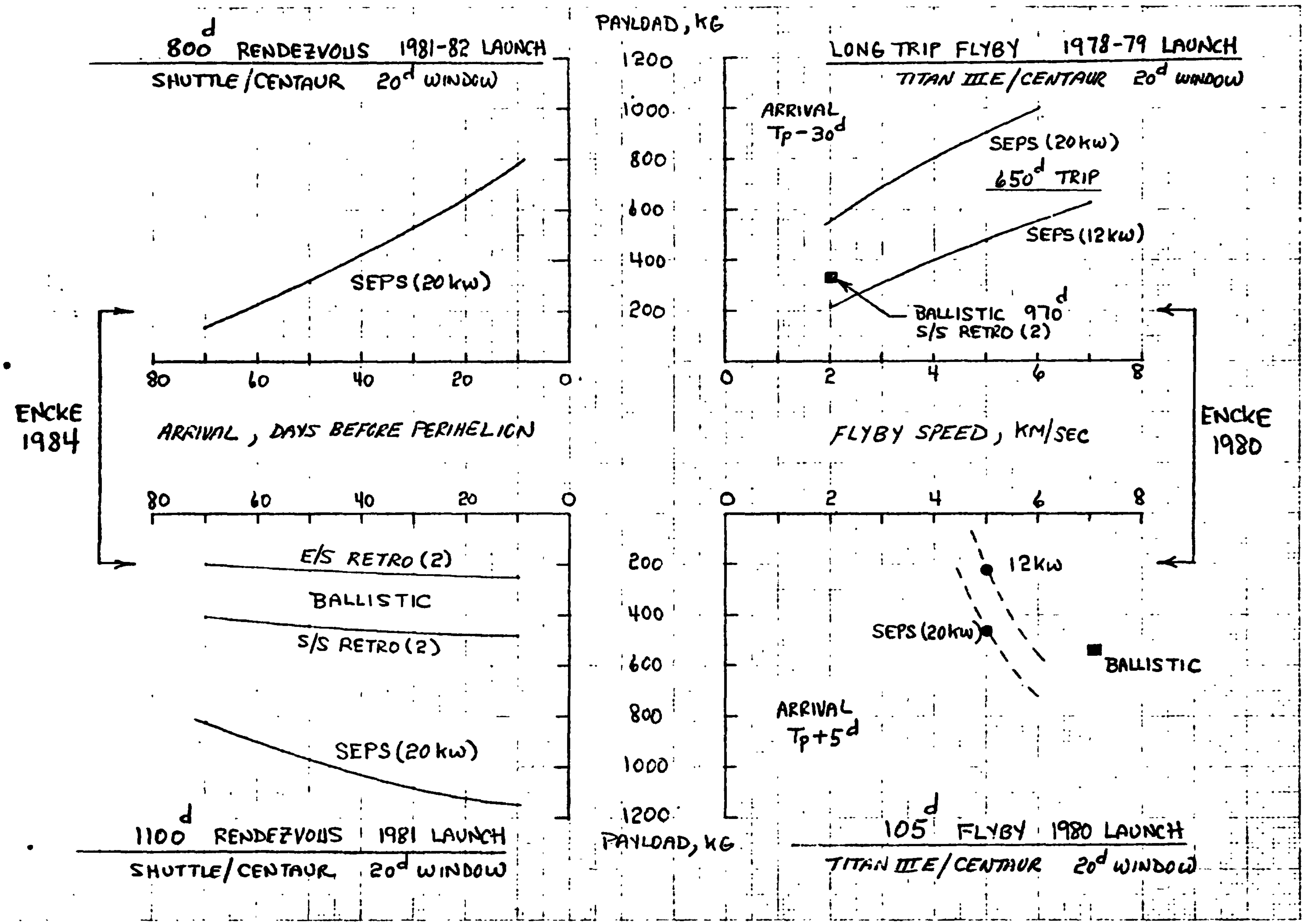
OCT. 31, 1973

1. SATURN MISSION PERFORMANCE
2. ENCKE MISSION PERFORMANCE
3. ENCKE MISSIONS (ALTERNATIVE ILLUSTRATION)



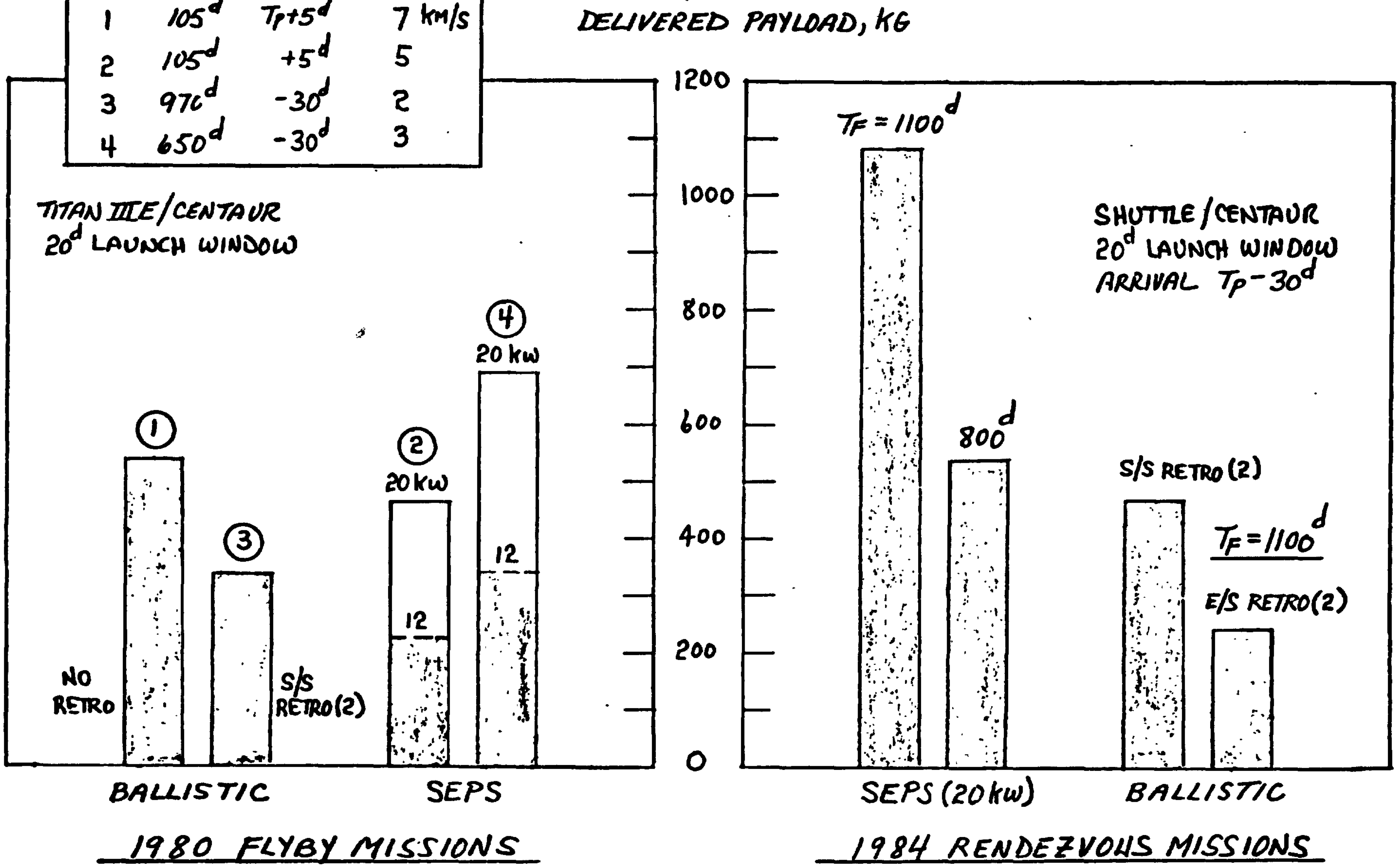
COMPARISON OF SOLAR ELECTRIC AND BALLISTIC FLIGHT MODES FOR SATURN ORBITER MISSIONS (1985 LAUNCH)

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC FLIGHT MODES FOR MISSIONS TO COMET ENCKE

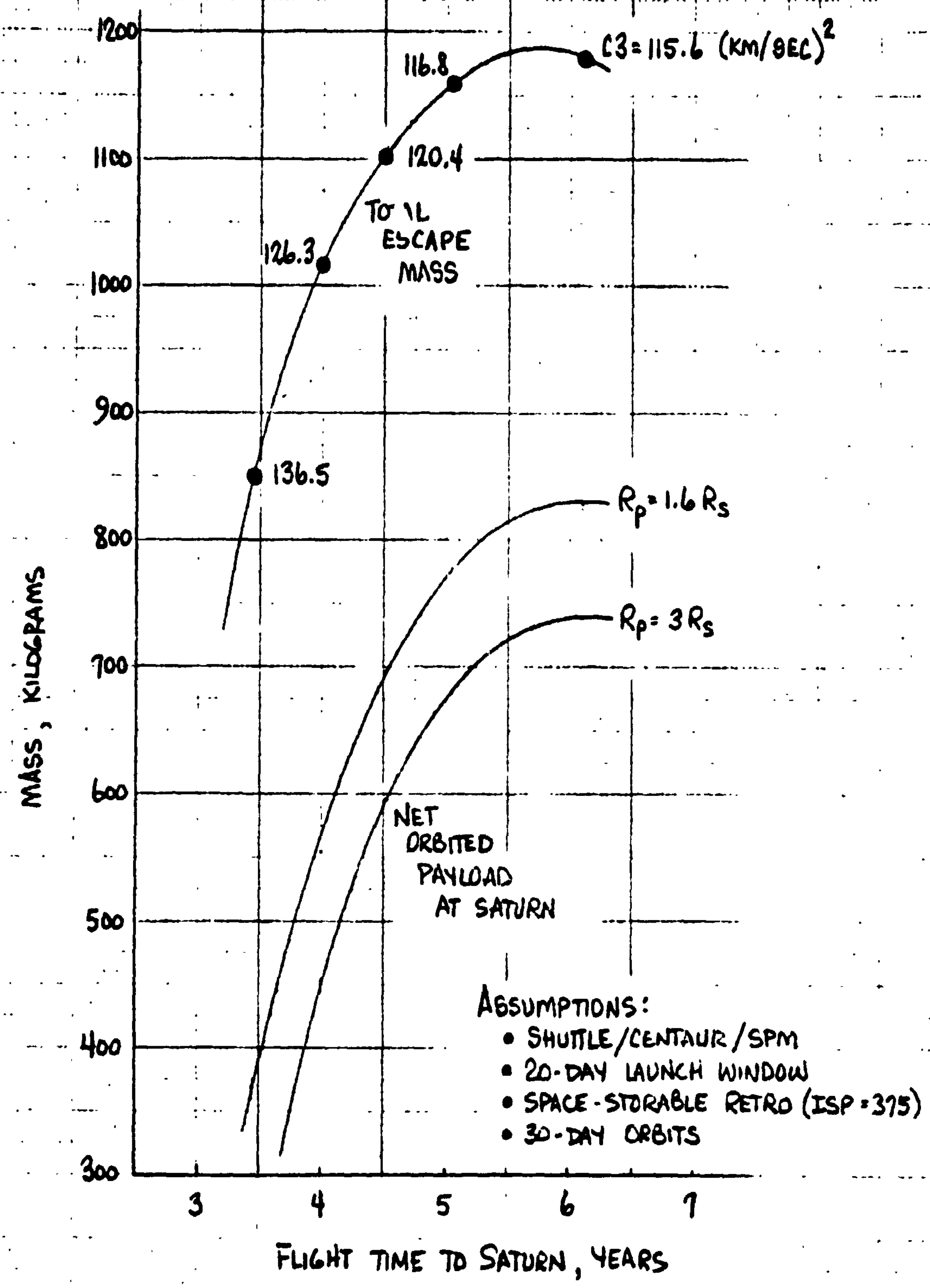


COMPARISON OF SOLAR ELECTRIC AND BALLISTIC FLIGHT MODES FOR MISSIONS TO COMET ENCKE.

MODE	T_F	ARRIVAL	SPEED
1	105^d	T_P+5^d	7 km/s
2	105^d	$+5^d$	5
3	$97c^d$	-30^d	2
4	650^d	-30^d	3



1985 SATURN ORBITER PERFORMANCE CURVES



1985 SATURN ORBITER PERFORMANCE CHARACTERISTICS

JCN/SAI

OOS TUG EVALUATION

PLANETARY MISSIONS ASSESSMENT

OF

ORBIT. TO. ORBIT. STAGE (OOS)

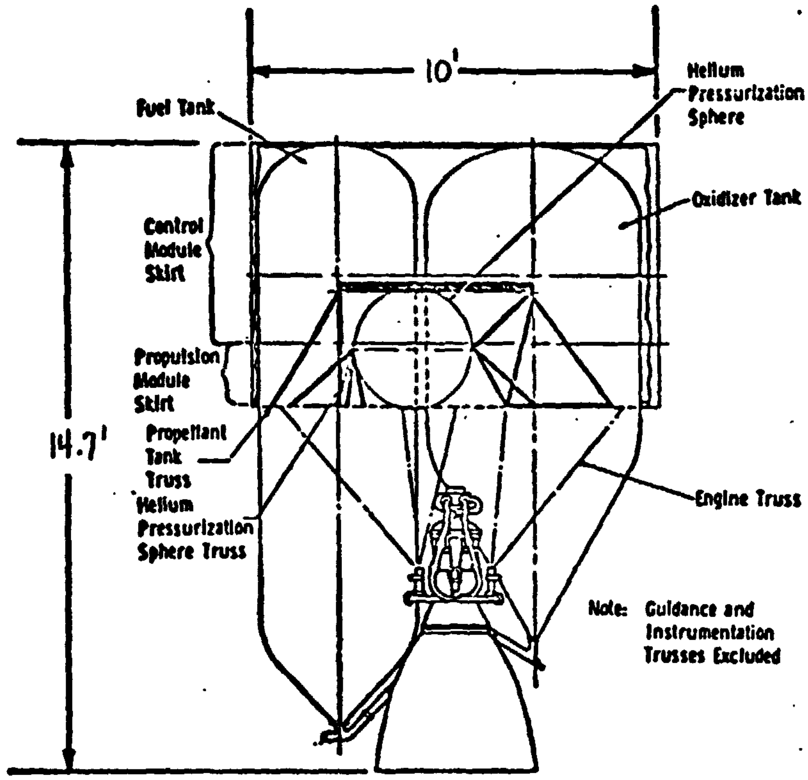
OPTIONS

PLANETARY PROGRAMS DIVISION
CODE SL/OSS
NASA HEADQUARTERS
WASHINGTON, D. C.

10 DECEMBER 1973

ASSESSMENT GUIDELINES

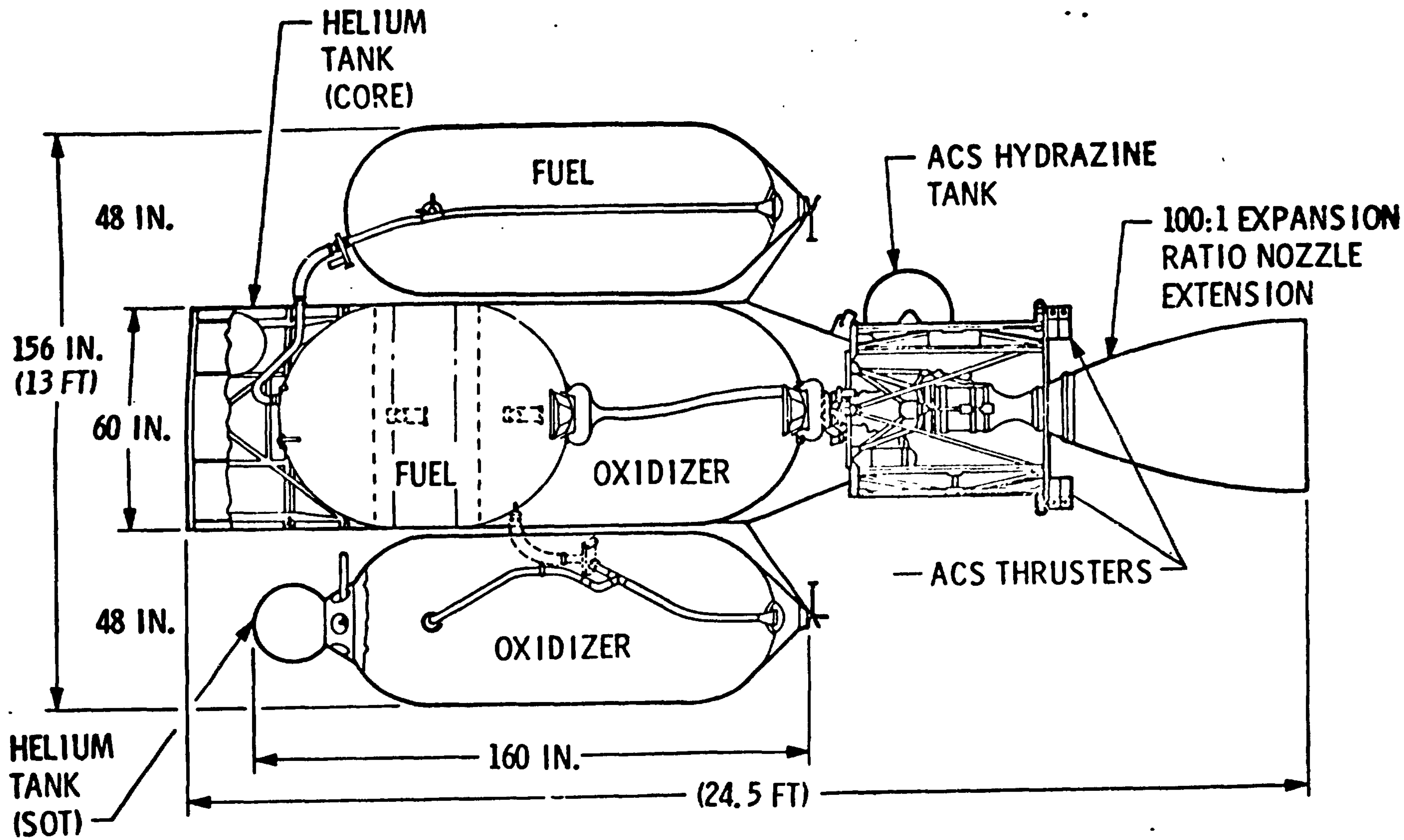
- ASSESS ON-ORBIT-STAGE (OOS) PLANETARY MISSION PERFORMANCE CAPABILITY DURING "TRANSITION PERIOD (1981-84) TO THE SPACE SHUTTLE.
- USE LATEST PLANETARY MISSION MODEL WITH UPDATED MISSION DEFINITIONS FROM RECENT ADVANCED STUDIES WHERE APPLICABLE.
- CONSIDER TRANSTAGE, AGENA AND CENTAUR DOD OOS CANDIDATES.
- CONSIDER BOTH EXPENDABLE AND REUSABLE OOS FLIGHT MODES.
- USE 21 KW SEP STAGE TO AUGMENT ESCAPE PERFORMANCE WHEN NECESSARY.
- EMPLOY MSFC AND CODE SV ADJUSTED COST ESTIMATES IN ASSESSMENT OF COST EFFECTIVENESS.



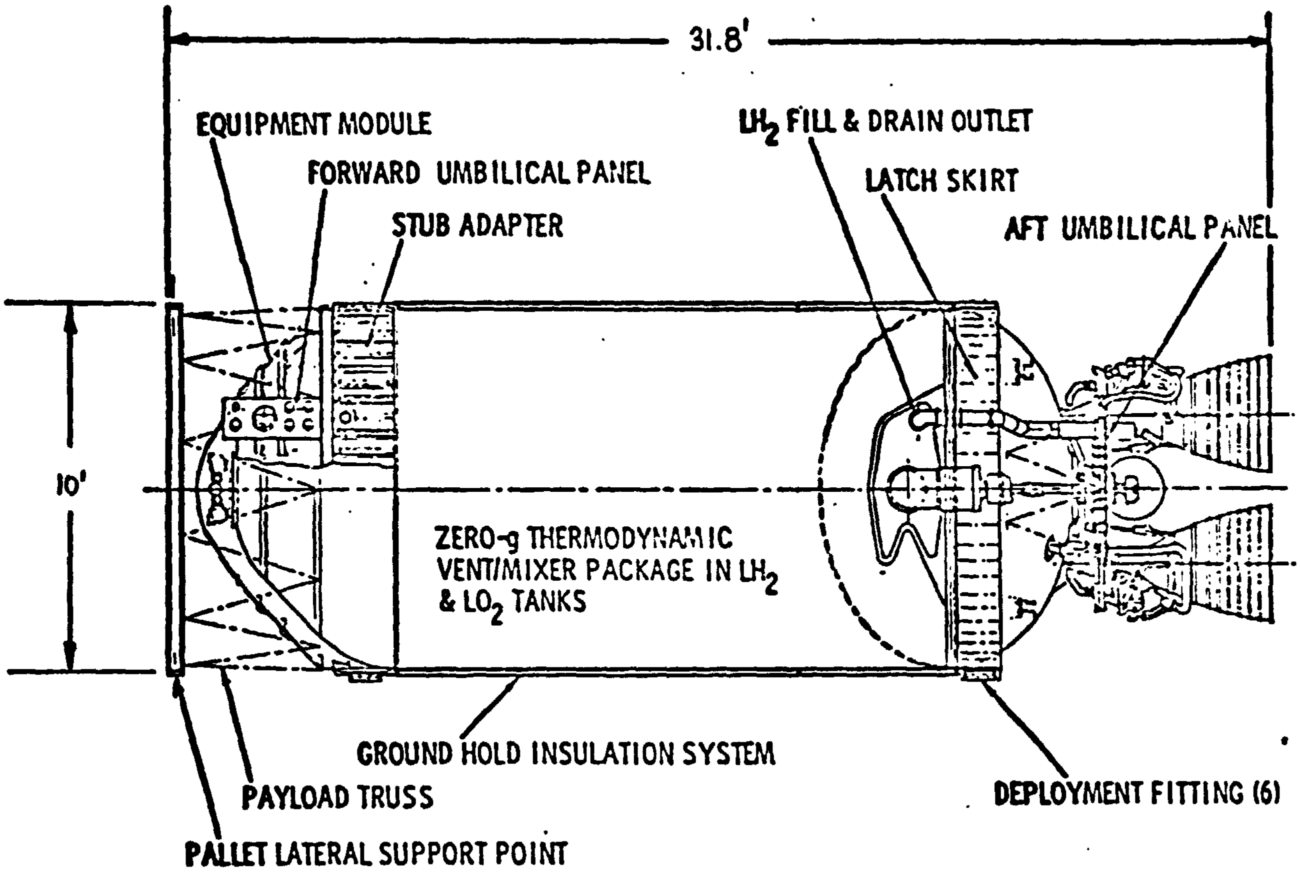
BASELINE SHUTTLE TRANSTAGE

REUSABLE OOS TRANSTAGE:

- 5.2 FEET LONGER
- SAME WIDTH
- 38 % MORE PROPELLANT



GROWTH AGENA



Baseline Shuttle Centaur

Reusable OOS Centaur:

- 3.8 feet shorter
- 4.5 feet wider
- 50% more propellant

OOS Performance Characteristics

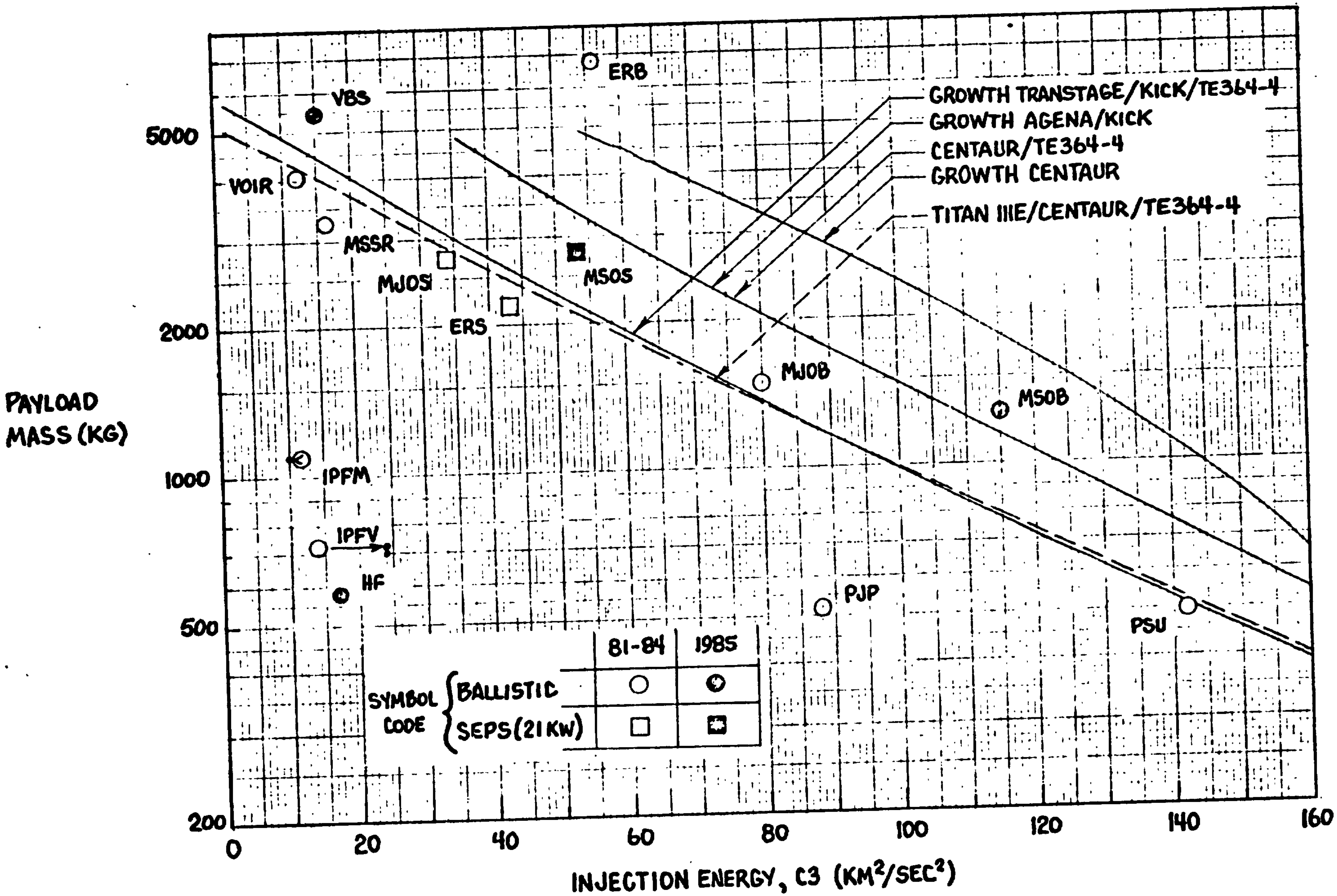
Stage	Ignition Wt. (LBM)	Mass Fraction	ISP (SEC)	Length (FT)	Width (FT)
Growth Transtage	37027	.86	311	19.5	10.0
Growth Agena	59106	.94	324	24.5	13.0
Baseline Shuttle Centaur	35642	.815	440	31.8	10.0
Growth Centaur	51330	.815	440	28.0	14.5
Kick Stages for:					
Centaur (TE 364-4)	2410	.955	284	5.9	3.3
Growth Transtage	4735	.84	289	6.2	NA
G. Agena, G. Centaur	10000	.84	288	6.2	NA

COST ESTIMATES

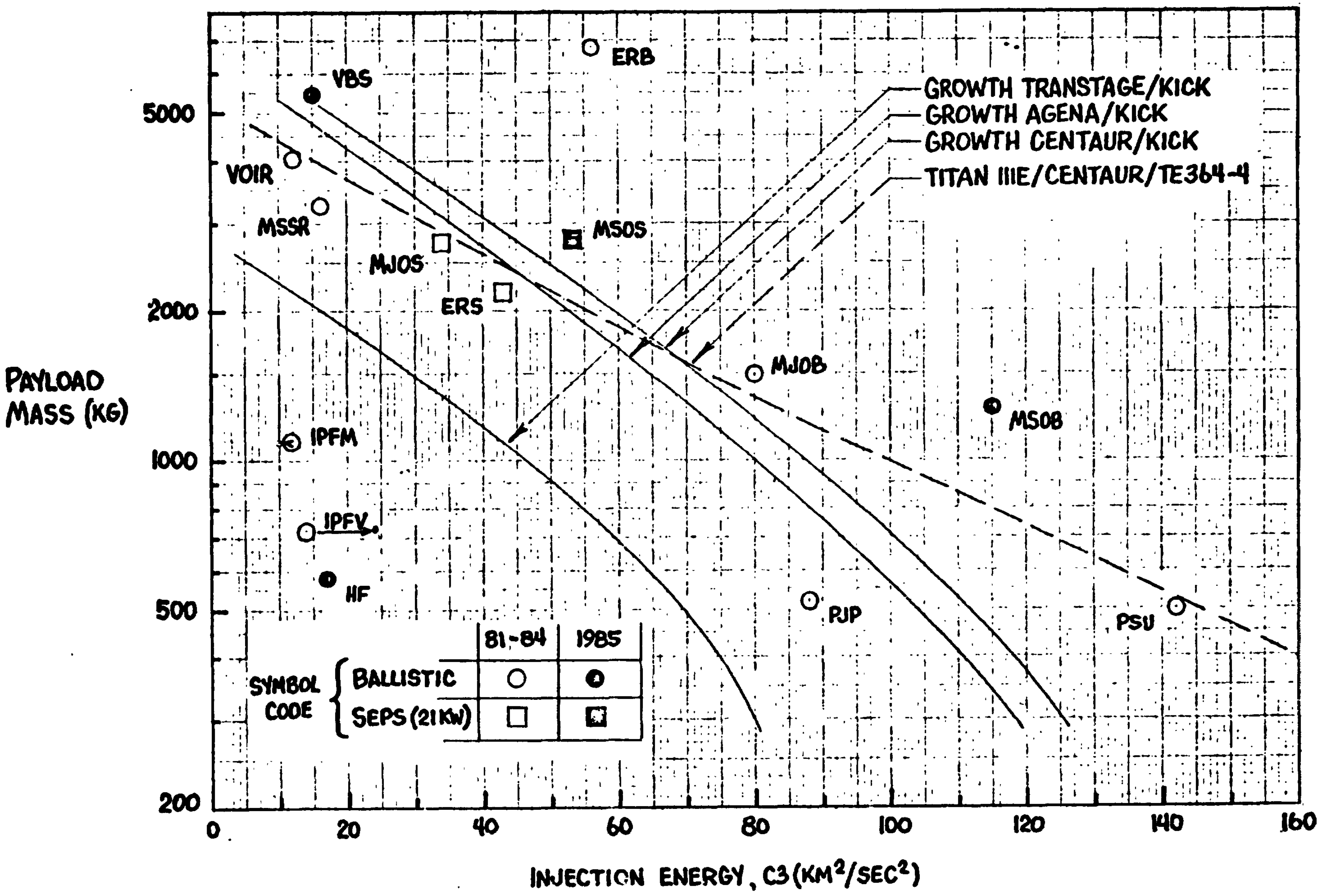
(MILLIONS OF 1973 DOLLARS)

STAGE	DDT & E	FLIGHT COST	
		EXPEND	REUSE
TITAN IIIE/CENTAUR/TE364-4	–	26.0	–
GROWTH TRANSTAGE*	94	6.5	1.1
GROWTH AGENA*	121	9.6	1.1
BASELINE CENTAUR/TE364-4	34	6.0	–
GROWTH CENTAUR*	113	10.8	1.1
KICK (4735 LBM)	30	1.2	–
KICK (10000 LBM)	30	1.4	–
SEPS (21 KW)	120	20.0	–

* REUSABLE CAPABILITY



PERFORMANCE CAPABILITY OF OOS CANDIDATES (EXPENDED)



PERFORMANCE CAPABILITY OF OOS CANDIDATES (REUSABLE)

SOME COST COMPARISONS[1]

OPTION NO.	OOS OPTION	NO. OF TITAN/CENTAUR LAUNCHES	NO. OF OOS FLTS.		OOS-KICK DDT&E	TOTAL LAUNCH COSTS[2]
			REUSE	EXPEND		
1	GROWTH TRANSTAGE/KICK	9	3	2	124M	306.3M
2	GROWTH TRANSTAGE/KICK	5	3	6	124M	273.1M
3	GROWTH TRANSTAGE/KICK	0	3	11	124M	231.6M
4	GROWTH CENTAUR/KICK	9	5	0	143M (+19)[3]	292.3M (-14)
5	GROWTH CENTAUR/KICK	5	9	0	143M (+19)	235.5M (-38)
6	GROWTH CENTAUR/KICK	0	11	3	143M (+19)	192.9M (-39)

1. ASSUMES A TOTAL OF 14 PLANETARY LAUNCHES DURING "TRANSITION PERIOD" (1981-84).
2. INCLUDES $10M PER SHUTTLE LAUNCH TO ORBIT WHERE APPLICABLE.
3. VALUES IN ( ) ARE COST DIFFERENCE COMPARED TO RESPECTIVE GROWTH TRANSTAGE OPTION.

CONCLUSIONS

- SOLAR ELECTRIC PROPULSION IS REQUIRED ON ONLY ONE MISSION - 1981 ENCKE RENDEZVOUS.
- REUSABLE OOS CAPABILITY CAN BE A SIGNIFICANT FACTOR IN TOTAL PLANETARY LAUNCH COSTS FOR "TRANSITION PERIOD".
- THE MORE OOS FLIGHTS SUBSTITUTED FOR TITAN IIIE/CENTAUR LAUNCHES, THE LOWER THE TOTAL LAUNCH COST.
- IT ALWAYS COST LESS TO LAUNCH THE SAME NUMBER OF PLANETARY MISSIONS WITH THE GROWTH CENTAUR OOS THAN THE GROWTH TRANSTAGE OOS BECAUSE OF GREATER CENTAUR REUSABILITY.
- GROWTH CENTAUR OOS HAS THE BEST PERFORMANCE MARGIN (USING BOTH REUSABLE AND EXPENDABLE MODES); THERE ARE NO "TIGHT" MISSIONS.
- GROWTH CENTAUR OOS WOULD PERMIT ONE OR TWO YEARS SLIPPAGE IN DEVELOPMENT OF THE HIGH TECHNOLOGY TUG WITHOUT DISRUPTING THE PLANETARY MISSION PLAN.

BAI LISTIC RENDEZVOUS WITH ENCKE 81/82

Ballistic Encke/81 Rendezvous

Preliminary analysis of the use of gravity assist to reduce energy requirements of a ballistic multi-impulse Encke/81 rendezvous mission has failed to turn up any positive results. Neither Jupiter nor Mars (see Figure) are properly situated for a useful swingby. It is doubtful that Venus would be of any interest due to its orbital motion relative to the transfer trajectory.

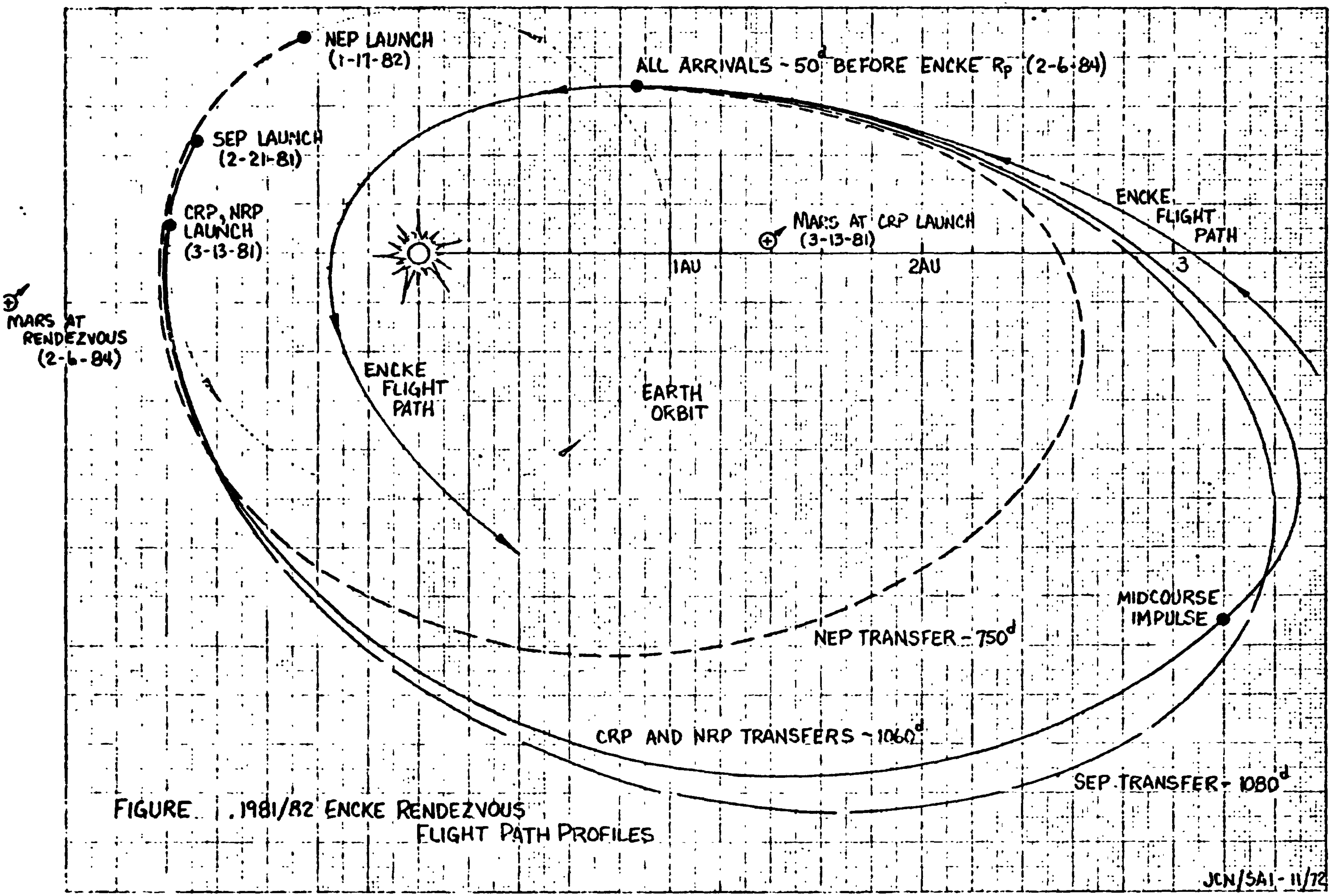


FIGURE . 1981/82 ENCKE RENDEZVOUS FLIGHT PATH PROFILES

COMET ENCKE 80 FLYBY - ASTEROID RENDEZVOUS MISSION

Comet Encke Flyby - Asteroid Rendezvous Mission

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Introduction

The scientific interest in comet and asteroid missions has been growing steadily over the past several years. This is due, in large part, to the activities of the NASA-sponsored science working groups and study panels.[1-3] The discovery this year of Comet Kohoutek and the flurry of activity focused on observing this comet has certainly raised the level of interest. A final factor is the advent of advanced propulsion technology, particularly solar electric propulsion (SEP), which will make possible future rendezvous, docking and even sample-return missions to small-body targets.

The current consensus among scientists and mission planners is that Comet Encke will be the primary target of early cometary exploration in the 1980 decade.[4] A two-mission sequence is planned which encompasses a flyby of Encke at its 1980 apparition (perihelion passage) to be followed by a rendezvous in 1984. Several preliminary design studies of the flyby mission are now underway. These studies are expected to provide the necessary tradeoff data from which NASA can make a more definitive selection of mission/spacecraft mode and science payload should the flight project be approved. The three principal mission modes under consideration are: 1) a short (90 day) ballistic transfer utilizing a modified Helios space-

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craft with encounter near Encke's perihelion at a flyby speed of 7-9 km/sec; 2) a short ballistic transfer utilizing a modified Pioneer-Venus spacecraft with encounter 16-26 days before perihelion at a flyby speed of 18-26 km/sec and possibly retargeted after Encke encounter to flyby the asteroid Geographos; and 3) a long (650 days) low-thrust transfer utilizing a 10-15 kw SEP spacecraft with encounter 20-30 days before perihelion at a slow flyby speed of 4-5 km/sec and possibly pre-targeted to flyby an asteroid prior to Encke encounter.

Attractive features of the SEP mission mode include the probable enhancement of science value due to the slow flyby speed, and the operational flight test of the SEP system as a precursor to the follow-on rendezvous mission to Encke. The main drawback is the inherently higher risk of mission failure associated with the new SEP technology. Furthermore, programmatical constraints could force a delayed project start putting the necessary Jan-Mar 1979 launch period in jeopardy. If this should prove to be the case, are there viable alternatives for a SEP spacecraft launched on a 1980 Encke flyby mission?

The purpose of this paper is to describe one such alternative. Basically, it is a multi-target mission mode which utilizes the SEP capability to rendezvous with an asteroid after the encounter with Encke. We could define this mode as a "no-risk" Encke flyby mission relative to SEP technology. Launched in mid-1980, the earth-Encke transfer is all-ballistic, and SEP operation begins after comet encounter and is relied upon only to accomplish the secondary target objectives. The following discussion is based on an exploratory analysis and is therefore limited in scope to a description of trajectory profile and spacecraft mass characteristics.

Results

Encke has an orbital period of 3.3 years, is inclined 11.9^{o} to the ecliptic plane, and passes through a perihelion distance of 0.34 AU on Dec. 6, 1980. The launch period for short ballistic transfers lies in August of 1980 near the ascending nodal longitude of Encke. As the encounter date varies between Nov. 6 and Dec. 6, launch energy C_3 increases from 42 to 97 km^2/sec^2, while flyby speed decreases from 27 to 7 km/sec. Such transfer arcs have a 1 AU aphelion distance, a perihelion distance between 0.73 and 0.34 AU, and are inclined about 12^{o} to the ecliptic. It is this type of orbit which must be reshaped to intercept a second target of opportunity after Encke encounter.

Geometrical considerations lead one to the conclusion that the asteroid target should be chosen either from the Amor group (Mars crossing orbits) or the Apollo group (earth crossing orbits). This choice tends to ensure flight times which are not excessively long and minimizes the propulsion energy needed to achieve rendezvous conditions. Upon examining the time-position characteristics of particular bodies in these groups, two asteroid were selected for investigation: Eros (433) and Geographos (1620). The orbit of Eros is inclined 10.8^{o} and has perihelion and aphelion distances of 1.13 and 1.78 AU. Similar data for Geographos are 13.3^{o}, 0.83 AU and 1.66 AU. Both bodies appear to have small characteristic dimensions (2-35 km), are elongated in shape like a football or a cigar, and have rotational periods on the order of several hours.

Figure 1 illustrates a typical trajectory profile (ecliptic plane projection) for the 1980 Encke flyby - Eros rendezvous mission. Launched on Aug. 18 with escape energy $C_3 = 60\ km^2/sec^2$, the spacecraft encounters Encke 20 days prior to its perihelion passage at a heliocentric distance of

0.6 AU and a relative flyby velocity of 21 km/sec. Earth is at 54° longitude at this time and is therefore in a good position for communications and correlation of ground-based and spacecraft science measurements. The SEP thrust program initiated after Encke flyby shapes the subsequent trajectory once around the sun for rendezvous with Eros on Jan. 10, 1982. Note again that earth is in a very favorable position at the time of rendezvous operations. Typical thrust on-time is 350-400 days over the 420-day second-leg transfer to Eros. The interspersed coast periods, particularly near rendezvous, will aid the attainment of high navigation accuracy. Also, it should be noted that the optimum thrust direction angle relative to the sun remains close to 90° during the entire flight; this implies a desired simplification in thrust vector control mechanization relative to solar array pointing requirements.

Figure 2 shows a typical trajectory profile for the Encke flyby - Geographos rendezvous mission. The earth-Encke ballistic transfer is the same as before. In this case the asteroid's time-position characteristic is such that the SEP spacecraft must "bide its time", setting up the rendezvous conditions over nearly 2 revolutions around the sun. The Encke-Geographos flight time is 490 days with rendezvous occurring on Mar. 21, 1982 after Geographos has passed through perihelion and is at 1.2 AU from the sun. Unfortunately, the earth is in near-conjunction during this time which means that ground-based communications and observation geometry is relatively unfavorable compared to the Eros mission. Longer coast periods can be specified for the Geographos mission with a typical thrust on-time of 250-300 days. Thrust pointing relative to the solar direction varies over a wider range, 55° - 120°.

As it turns out the payload delivery capability of SEP is nearly the same for either asteroid target. Performance results are summarized in Figure 3 where the Titan IIIE/Centaur/BII launch vehicle and a 11 kw SEP powerplant are assumed. The initial mass corresponds to the maximum launch vehicle performance as determined by the ballistic C_3 requirements, less a 10% penalty to account for a launch window of about 2 weeks. With the thrust subsystem operating at a specific impulse of 3000 sec and a total efficiency of 64%, the mercury propellant requirement varies from 370 kg to 465 kg over the 10-day Encke encounter window shown. The propulsion system is comprised of the solar arrays, power conditioners, thrusters and thrust vector control mechanisms, and is estimated to weigh 330 kg for the 11 kw system. Delivered "payload" or net spacecraft mass is given by the lower curve in Figure 3; this would be comprised of the science experiments ($\sim$ 60 kg) and the non-SEP functional support subsystems such as communications, data handling, thermal control, integrating structure, etc. One possible vehicle configuration would have the SEP Module attached to a 3-axis stabilized spacecraft based on Mariner and/or Viking technology. A net mass requirement of about 600 kg can be expected. This effectively constrains the Encke encounter date to be no later than 17 days before perihelion with a resultant minimum flyby speed of about 19 km/sec. It is important to note that the Titan/Centaur launch vehicle may be utilized without the BII kick stage since the C_3 requirement is sufficiently low in this region of constraint.

Increased payload performance would be possible if the Shuttle/Centaur vehicle is available for a 1980 launch. This is indicated by the performance map for the Encke-Eros mission shown in Figure 4. Net mass curves are linear functions of SEP power rating assuming a constant value of propulsion system specific mass (α is taken as 30 kg/kw). Optimum SEP power is 10-11 kw for the Titan/Centaur launch vehicle as shown by its maximum performance constraint boundary. In the case of Shuttle, optimum power is 16-18 kw which yields a net mass increase of 60-70 percent. Hence, Shuttle availability would provide a good measure of payload design margin and would allow later Encke encounters up to about 12 days before perihelion.

The pertinent conclusions from this analysis are: 1) an attractive multi-target mission alternative exists for Encke 1980 exploration; 2) SEP technology would be employed, at virtually no risk to cometary objectives, to rendezvous with an asteroid after Encke encounter; 3) of the two asteroid targets studies, Eros offers the better mission profile; 4) this mission could be the maiden SEP voyage replacing the proposed SEP slow flyby if its earlier launch date should prove to be programmatically impossible; 5) in any event, many future opportunities should exist for comet flyby - asteroid rendezvous missions (e.g. Halley 1986) which are uniquely suited to SEP capabilities. Other multi-target asteroid flyby concepts have been proposed elsewhere - rendezvous is much preferred for bodies of such small dimension. Finally, it appears that the proposed mission concept warrants further detailed analysis to verify its design and cost feasibility.

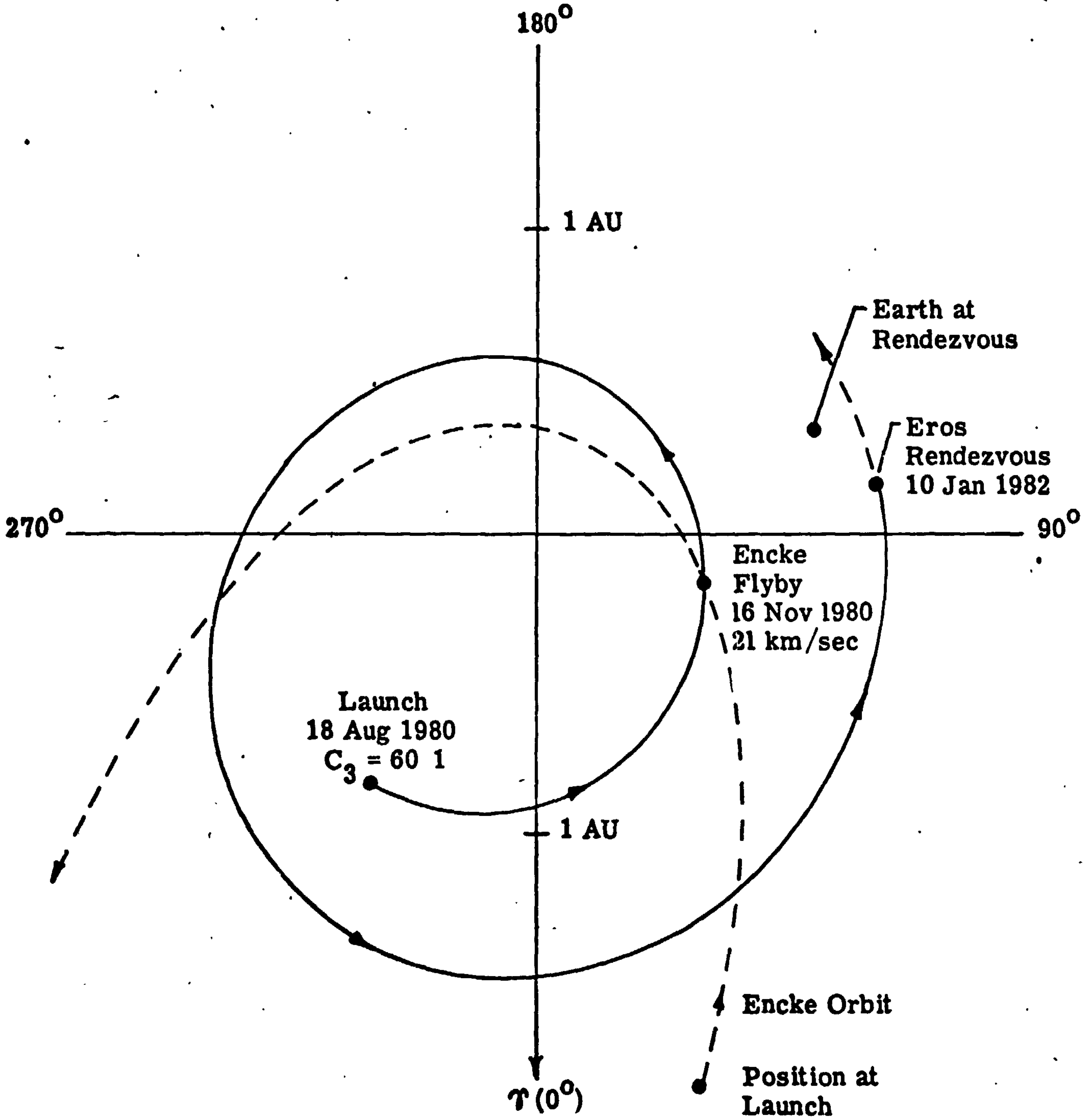
References

1. Roberts, D. L. (Ed), "Proceedings of the Cometary Science Working Group", Yerkes Observatory, June 1971.

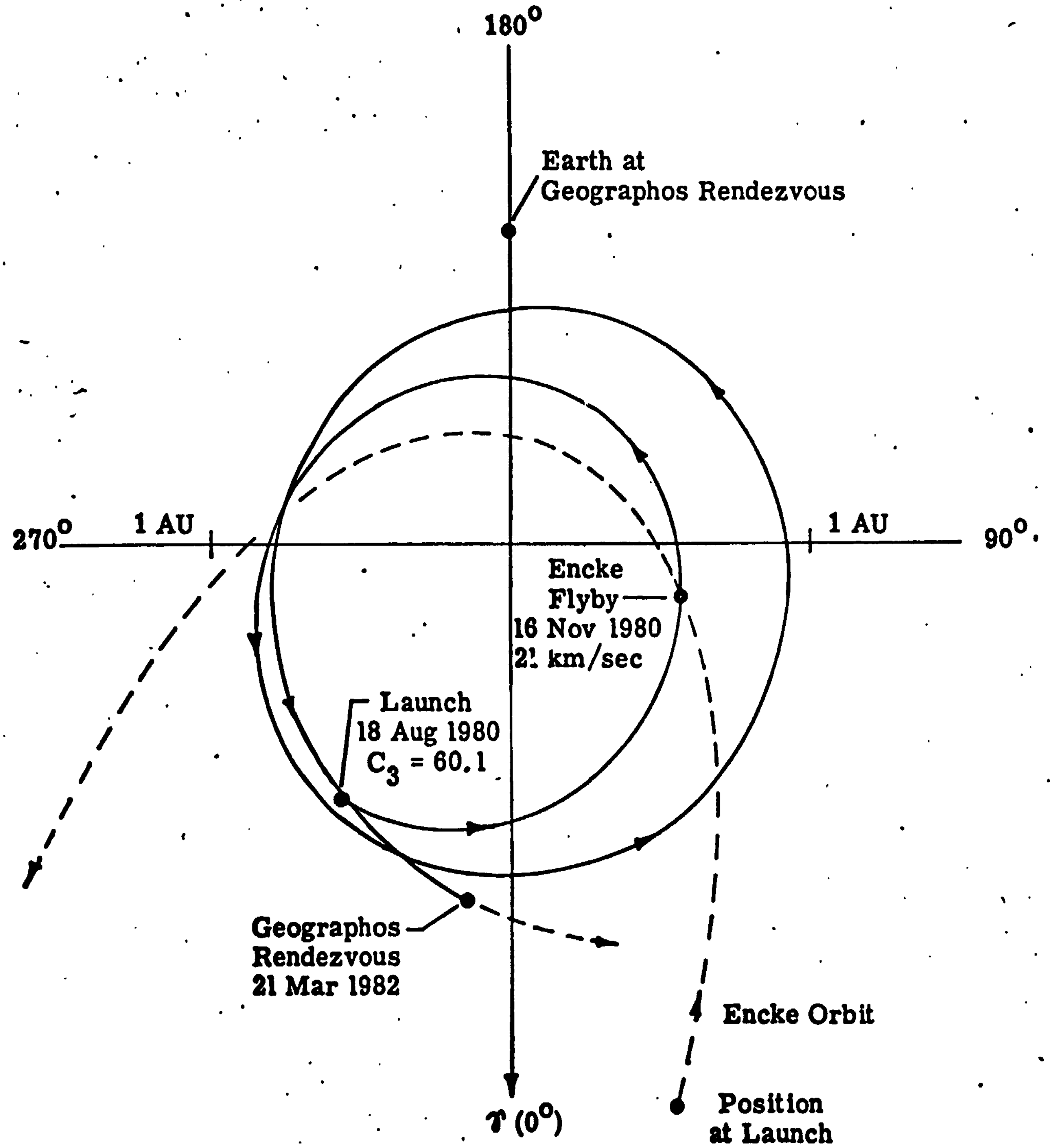
2. "Comets and Asteroids" A Strategy for Exploration", Report of the Comet and Asteroid Mission Study Panel, NASA TMX-64677, May 1972.

3. "The 1973 Report and Recommendations of the NASA Science Advisory Committee of Comets and Asteroids", NASA TMX-71917.

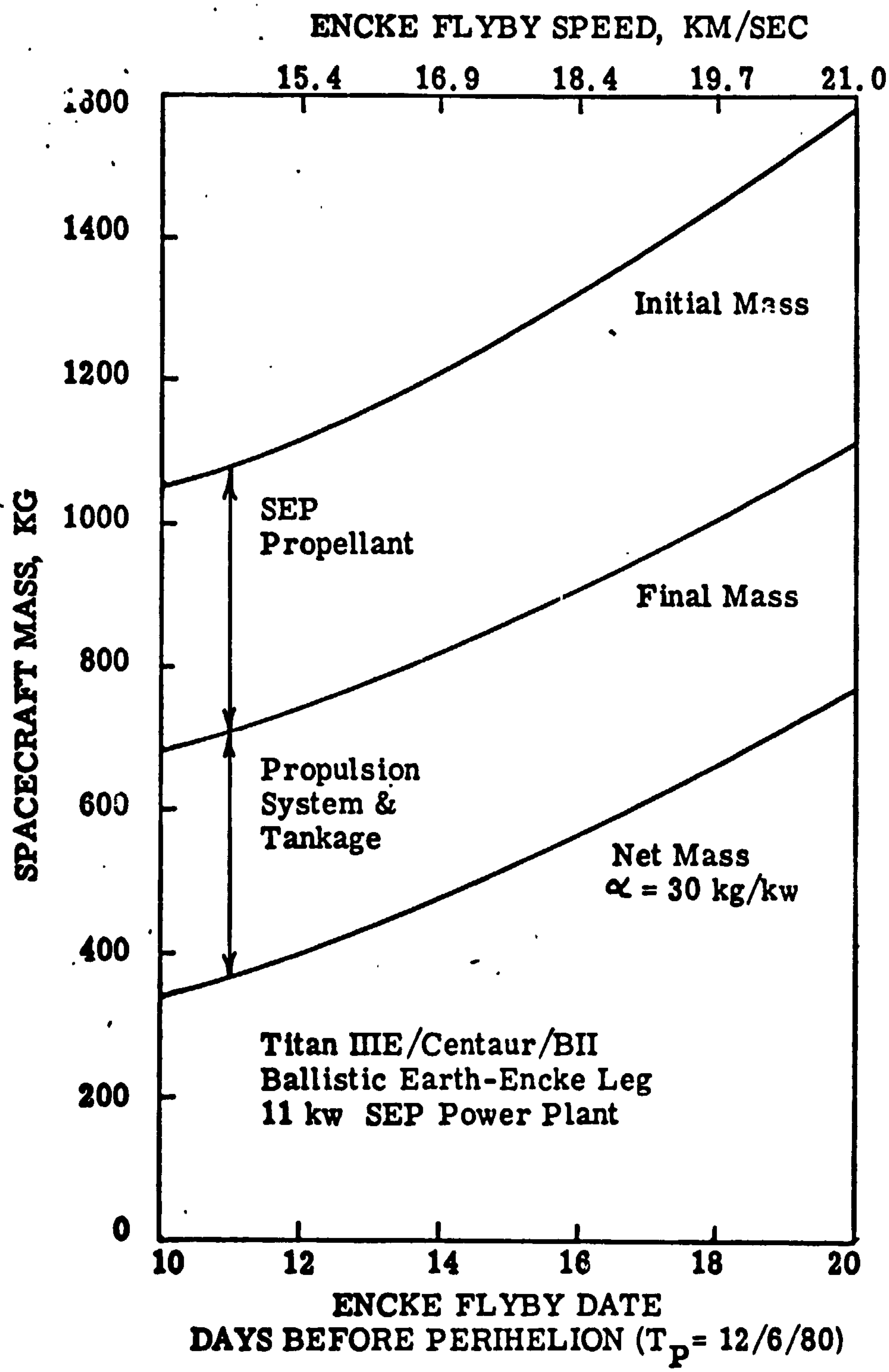
4. Atkins, K. L. and Moore, J. W., "Cometary Exploration: A Case for Encke", AIAA Paper No. 73-596, presented at the Space Mission Planning and Execution Meeting, Denver, Col., July 10-12, 1973.



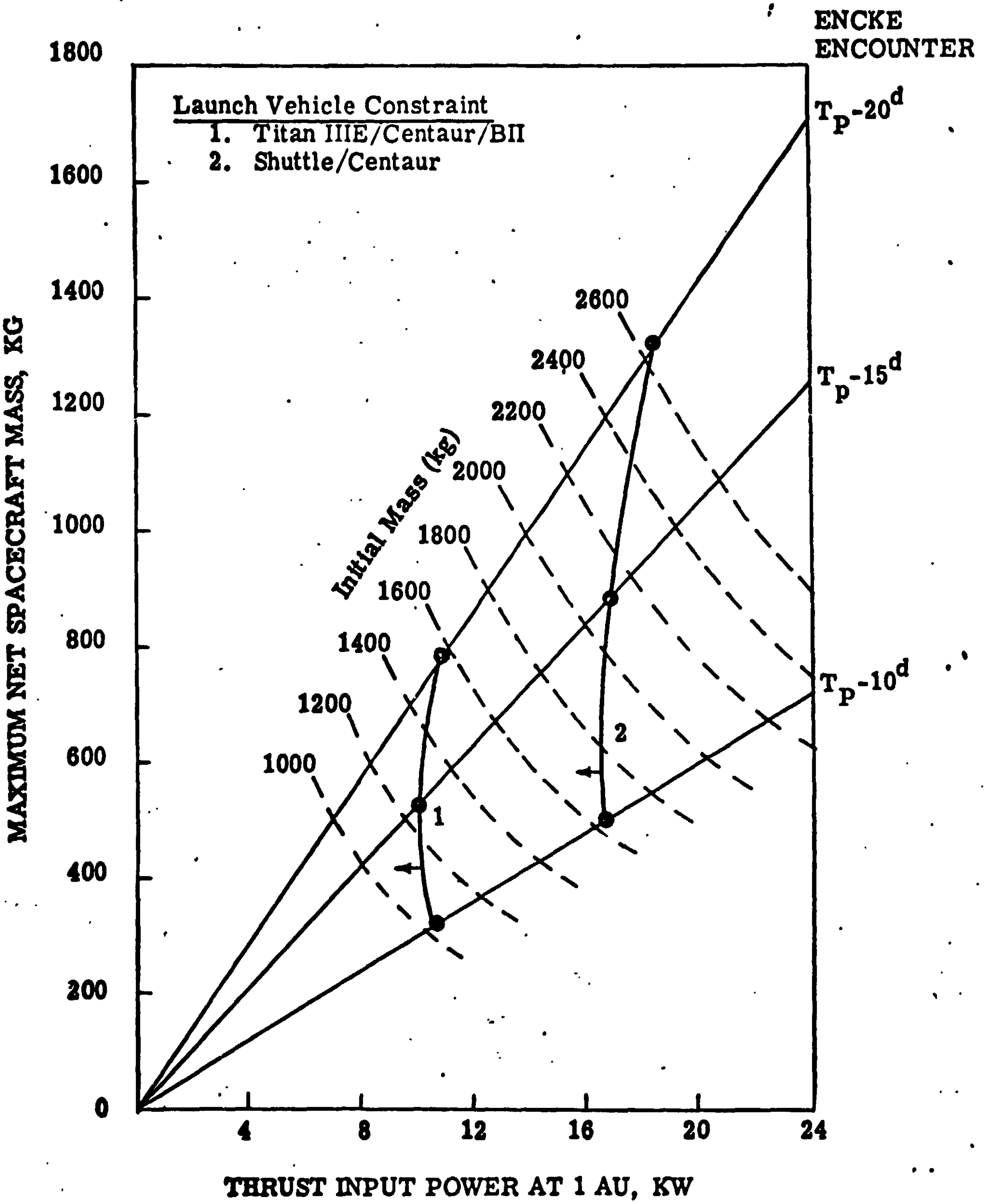
F - 1 Trajectory profile for 1980 Encke flyby - Eros rendezvous mission.



F - 2 Trajectory profile for 1980 Encke flyby - Geographos rendezvous mission.



F - 3 Spacecraft mass capability for Eros (or Geographos) rendezvous via solar electric propulsion.



F - 4 SEP performance map for 1980 Encke flyby - Eros rendezvous mission. (continuous thrust, α = 30 kg/kw)

PIONEER MARS 1979
MISSION OPTIONS

As part of its continual planning effort, the Planetary Programs Division of OSS/NASA has been developing a number of mission options for post-Viking/75 Mars exploration. For the two remaining Mars launch opportunities in this decade, i.e. 1977 and 1979, planning emphasis to date has been placed on derivatives of Viking/75 hardware. NASA's recent commitments to the development of the Space Shuttle in this same time frame could, however, reduce resources to a point where a follow-on Viking mission might not be possible until the early 1980's. If this were to happen, rather than completely abandoning Mars opportunities in the late 1970's, OSS/NASA would like to have several lower cost mission concepts available for consideration as alternatives.

The purpose of this study was to conduct a preliminary investigation of lower cost (<$100M) Mars missions which perform useful exploration objectives after the Viking/75 mission. As a study guideline, it was assumed that significant cost savings would be realized by utilizing Pioneer hardware currently being developed for a pair of 1978 Venus missions. This in turn led to the additional constraint of a 1979 launch with the Atlas/Centaur launch vehicle which has been designated for the Pioneer Venus missions.

Selection of science-effective Pioneer mission concepts which would follow the Viking/75 mission without competing with future Viking missions in the early 1980's was accomplished by a process of elimination. Flyby concepts, e.g. a probe/relay bus, a remote sensor platform, or an atmospheric aeronomy platform, were all rejected because of the inadequate sampling time available considering the advanced state of Mars exploration. Low cost atmospheric entry probes and rough landers were rejected because their science potential is largely redundant to Viking/75 objectives. Two concepts, using an orbiter bus platform, were identified wh[illegible] both good science potential and mission simplicity indica[illegible] lower cost. These are: a) an aeronomy/geology orbiter, and b) a remote sensing orbiter with a number of deployable surface penetrometers.

Mission A, the Aeronomy/Geology Orbiter, would perform in situ aeronomy measurements in the Martian ionosphere by using low periapse altitude ($\approx$100 km) elliptical orbits. The low altitudes in the region of periapse also permit the inclusion of several remote sensing instruments capable of performing geologic surface mapping, e.g. a radar altimeter and a γ-ray spectrometer. Key mission parameters developed in this study are summarized in the Summary Table. Both the aeronomy and geology measurements would extend similar Viking entry/lander science data to a global scale. The trade-off for this capability is sterilization of the entire Pioneer orbiter spacecraft in order to meet Mars planetary quarantine requirements. Because the spacecraft passes through the upper atmosphere every orbit, its lifetime, even with periapse control, is only several years at best. The cost of this mission, excluding science, is estimated to be about \$31M (FY '74 dollars). This assumes the modification of an additional Pioneer Venus orbiter flight article, including sterilization, for a single launch in 1979. Suitable aeronomy instruments are readily available from many earth satellite programs, some of which have already been proposed for the Pioneer Venus orbiter mission in 1978. Appropriate remote sensing geology instruments are much more questionable, especially the γ-ray spectrometer, and could require significant development. Still, a total mission cost of \$40-50M dollars seems reasonable.

Mission B, the Remote Sensing/Penetrometer Orbiter would sequentially deploy a number of surface penetrometers to preselected impact sites distributed in either the northern or southern hemisphere of the planet. In addition to being a communications relay station between a deployed penetrometer and the earth, the orbiting bus could carry a complement of remote sensing instruments for orbital investigation of the Martian atmosphere and surface. Key mission parameters developed in this study are given in the Summary Table. A total of four sterilized penetrometers would be carried by a modified Pioneer Venus orbiter bus. These would be deployed one at a time from an elliptical polar orbit over a period of time as long as one Mars year. Each penetrometer would have its own deorbit motor and entry/descent system. Penetrometer design and descent velocity specification provide for a minimum penetration of 1 m in rock without destruction. During a 1-week surface lifetime each penetrometer would identify soil penetrability, search for subsurface water, and perform an elemental chemical analysis of the subsurface material at its impact site. The data collected from its instruments would be transmitted to the orbiter once each Mars day for relay back to earth. Between the four one-week penetrometer missions the orbiter could perform remote sensing measurements with its own science package. The factors of low cost, low power, low data rate, and high minimum altitudes (>1000 km) probably restrict these measurements to atmospheric studies with existing or slightly modified instruments. The scientific merit of such experiments in 1980 requires further study. The cost of this mission, excluding orbiter science, for a single 1979 launch is estimated to be about \$63M. This figure includes \$24M for the development

Summary Table

SELECTED PIONEER MARS MISSION CONCEPTS

- Mission A: Aeronomy/Geology Orbiter

  - o 50-70 kg science payload
  - o Aeronomy and surface geology science instrumentation
  - o 300-350 kg orbited payload
  - o $\geq$ 100 km periapse altitude
  - o 24 hour initial orbit period
  - o 45^{o} orbit inclination
  - o $\approx$One Mars year orbit lifetime
  - o Entire spacecraft sterilized

- Mission B: Remote Sensing Orbiter/Penetrometers

  - o 40-60 kg orbiter science payload
  - o Four impact penetrometers @ 40 kg each
  - o Penetrability, water detection, and soil chemistry impact science instrumentation
  - o 500-550 kg orbited payload
  - o 1000 km periapse altitude
  - o 24.6 hour controlled orbit
  - o 90^{o} orbit inclination
  - o >42 year orbit lifetime
  - o $\approx$One week penetrometer lifetime
  - o Penetrometers sterilized

and fabrication of four penetrometers (including penetrometer science), one flight spare and a PTM. Depending on the selected orbiter remote sensing experiments, total cost (excluding launch vehicle) for the Remote Sensing/Penetrometer Mission could have a range of $70-80M (FY '74 dollars).

This exploratory analysis has identified and outlined at least two 1979 Mars mission concepts, based on Pioneer Venus technology and hardware, which have the potential for performing relevant post-Viking/75 science at a cost of less than $100M. Mission A, the Aeronomy/Geology Orbiter, represents a minimum development/cost mission estimated at less than $50M. Yet the broad sampling of ionospheric composition and heat balance performed by this mission would greatly expand the data base from which scientists are trying to understand the evolution of the Martian atmosphere. Further, its potential for performing global geologic mapping from low altitude, gained by sterilizing the entire spacecraft, is not possible with the present Viking orbiter design.

Mission B, the Remote Sensing/Penetrometer Mission, is a somewhat more expensive mission, with in situ surface objectives, estimated at a cost of $70-80M. This mission requires the development of high impact ($\approx$150m/sec) penetrometers for which there exists an impressive history of earth-based experience. Pioneer Venus orbiter modifications would also be more significant than for Mission A. The science highlights of this mission are a) global exploration for subsurface water and b) establishment of a basis for extension of Viking Lander geologic data to global interpretations. The orbiter has the capability to perform continued non-imaging remote sensing studies of Mars from a polar orbit. The penetrometer concept also is a viable candidate for additional missions after 1979. Besides deploying the same penetrometers to more sites, there is the potential for a penetrometer/seismometer experiment pending development of a longer life ($\approx$90 day) power source.

It is important to point out that neither of these concepts should be considered feasible on the basis of this study. Many engineering questions exist for both concepts which require further study. Indeed, the actual Pioneer Venus Orbiter spacecraft design was not known at the time this analysis was performed. Undoubtedly there are solutions for each engineering problem which can be developed in a spacecraft systems study. The important question to be answered is: "How do these solutions change the definition and cost of the missions?"

It is equally important to note that the potential role of Pioneer-class Mars missions has not been thoroughly explored by a NASA science advisory group.[1] This potential should be refined for various post-Viking/75 Mars exploration scenarios as more and better definitions of Pioneer Mars mission concepts are developed.